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VOLUME I

TECHNICAL DRAFT STUDY REPORT FOR TOPEX SATELLITE OPTION STUDY

MARCH 11, 1982

JPL CONTRACT NO. 956200

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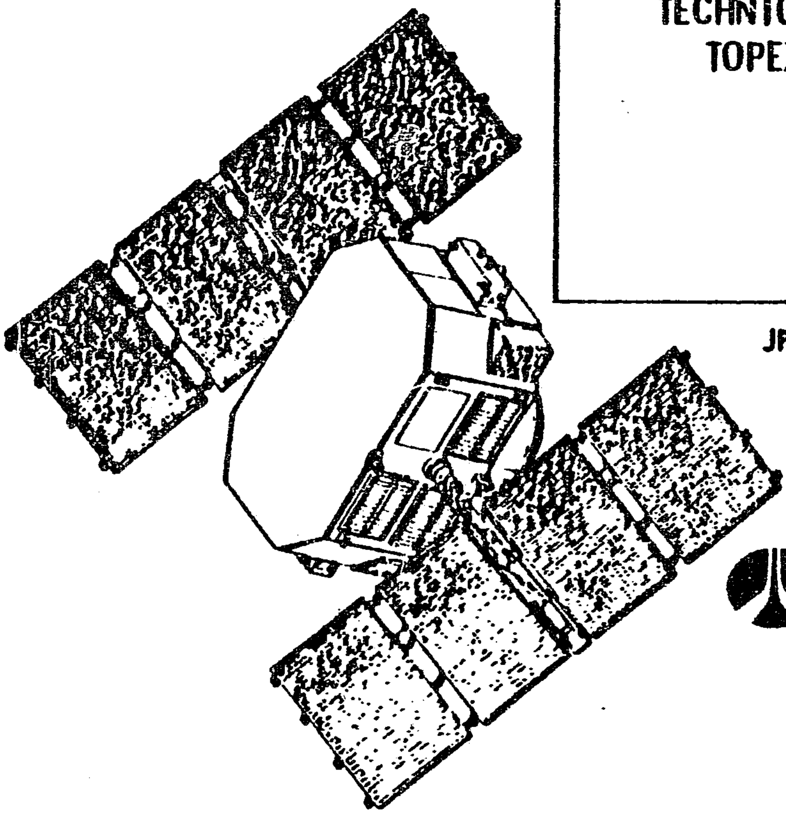

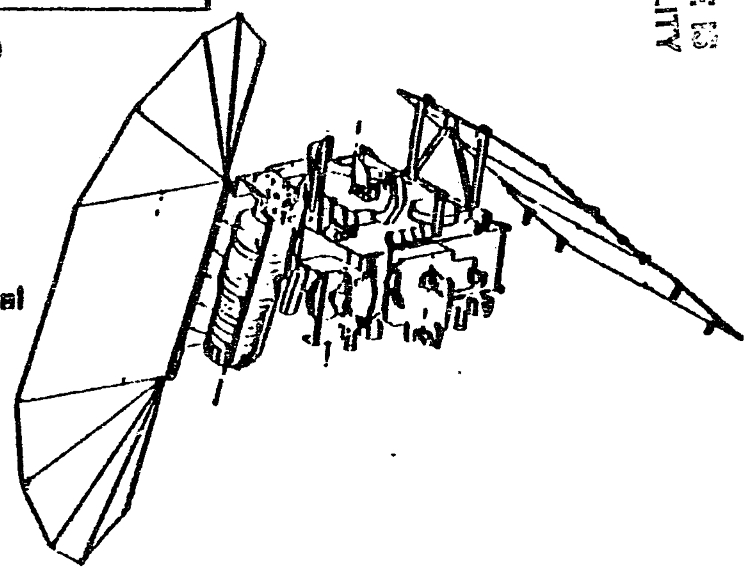

Rockwell International

Space Operations/Integration & Satellite Systems Division
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THE DYNAMIC OCEAN TOPOGRAPHY EXPERIMENT (TOPEX) STUDY REPORT

INTRODUCTION

This study was conducted for the Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California. The study considered the use of two (2) Rockwell-designed spacecraft for adaptation to the TOPEX mission, namely the P80-1 and the GPS Phase II.

The mission involved three (3) mission options, each option varying in payload definition, payload weight, orbital altitude and payload power requirements, as defined in Exhibit 1 of the Statement of Work in JPL Contract No. 956200.

The P80-1 spacecraft is an Air Force Space Test Program satellite which carries a number of payloads to an orbital altitude of 400 n.mi. at a minimum inclination of 72.5°, and which has an orbital life capability of three years. The GPS Phase II spacecraft is the operational satellite for the Global Positioning NAVSTAR navigation constellation provided for all-service (and commercial) use. Its predecessor, GPS Phase I, is the developmental satellite system and is presently orbiting six spacecraft all of which are still operating. A seventh is scheduled for launch in August, 1982 and three more are being built for launch in the near future. Most of the highly reliable hardware used in GPS I is being carried over for use in the operational GPS Phase II satellites.

The three mission options mentioned all share the following characteristics:

- Three-year mission with two-year extended option
- 10-day repeat circular orbit at 63.4° inclination, repeating within 1.0 kilometer
- An orbit eccentricity of <0.001
- Shuttle launch from the Western Launch Site to 150 n.mi. with 63.4o inclination
- Payload Operations Control Center (POCC) and support at JPL
- Telecommunications and operational orbit determination via TDRSS
- Altitude measurement within two centimeters
- Time tag resolution less than four μ seconds with rollover eight years
- FY84 project start (8-1-84) - late 1987 (11-87) launch

The two candidate spacecraft were studied with respect to their capability to meet the mission and TOPEX payload requirements in the following areas:

- Payload accommodations
- Attitude determination and control
- Command and data handling
- Telecommunications
- Ascent propulsion
- Launch vehicle compatibility

In addition, the following characteristics were derived either from the JPL Study Team Phase A Report, (Document No. 1633-1, dated 9-81) or from study activity derived internally. These include:

- Data downlinking is not required during eclipse
- Data downlinking periods are 22 minutes long
- A power margin of 10% is reserved for bus subsystems
- Orbital period is 112 minutes with a maximum eclipse period of 34.7 minutes
- Spacecraft RCS thrusters must be capable of conducting a minimum delt-V maneuver of 10 millimeters/second, with an accuracy of 1 millimeter/ second in track

Information on the Aerospace Support Equipment (ASE) required to carry, monitor and separate the two candidate spacecraft from the cargo bay of the Shuttle Orbiter has also been included and was used to show the system weight changes from lift-off to orbital operations.

Five separate tasks were assigned in the study and each task is defined at the beginning of the task response in the study organization. An up-front summary is included, followed by the task responses. At the end of the study, sections have been included which cover the ramifications of space system testing, ground support equipment, software, preliminary mission sequencing and program scheduling, and ending with some statements on conclusions and recommendations.

It should be noted that, although required modifcations of P80-1 and GPS II were identified and described, specific design analysis or trades were not performed in this study.

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SUMMARY

This summary does not include the cost elements. Costs will be submitted in a separate volume.

This cursory study indicates that either the P80-1 A.F. Space Test Program satellite, or the Global Positioning System (GPS) operational phase (Phase II) satellite design can meet the spacecraft system portion of the TOPEX mission requirements delineated in the Introduction of this study report or in the contract Statement of Work. Both spacecraft can meet the three-year mission without modification. The GPS II can meet the two-year extension option without modification but the P80-1 would probably require additional RCS propellant storage. Neither spacecraft's Telemetry, Tracking & Command (TT&C) subsystem can meet the NASA TDRSS/GSTDN telecommunications requirements; however, both spacecraft are easily adaptable to the NASA Command & Data Handling/ TDRSS Telecommunications Subsystem. Equipment can also be accommodated to allow on-board operational orbit determination via TDRSS RF.

Both spacecraft are capable of meeting an 8-1-84 start date and a launch in late 1987. The P80-1 mission will have been completed by the start date, and the GPS II spacecraft system qualification will be complete within four months after TOPEX go-ahead.

With respect to:

- Payload accommodations: Both spacecraft have adequate platform mounting room, both internally and externally, to mount all TOPEX payload components except for the Option 1 Radio Altimeter two-meter antenna on GPS II.
- Attitude determination & control: P80-1 can meet all TOPEX requirements with no modification; in fact, it's overdesigned. GPS II will require some modification for low earth orbit application.

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- Command & Data Handling: Requires elimination of both spacecraft's TT&C subsystems and the substitution of a NASA C&DH/TDRS subsystem.
- Telecommunications: both space spacecraft have the mounting space and electrical power to support the C&DH/TDRS subsystem. GPS II may require some modification to locate and mount the TDRS antenna.
- Payload accommodation: Adequate electrical power and thermal control can be provided by both spacecraft to accommodate all TOPEX payload requirements, both externally and internally to the spaceframe. Thermal control can be done passively.
- Off-the-shelf solid rocket motors are available to meet the perigee and apogee insertion ascent propulsion requirements of the TOPEX mission. Adaptation of different insertion stages to P80-1 is a simple matter; however, GPS II, a spin stabilized spacecraft, would require a modification to adapt any stage other than the PAM-D perigee insertion motor to the spin-table located in the launch cradle. The existing stages for the P80-1 might be trimmable to allow insertion and adequate plane change capability from "standard" Shuttle Orbiter launch inclinations to the TOPEX 63.4°.
- Launch vehicle compatibility: Neither of the two spacecraft were reviewed relative to launch on an expendable Delta booster; however, both spacecraft have been designed for a Shuttle launch and P80-1 even has its own dedicated launch cradle and separation support equipment which could be made available for the TOPEX mission. Both spacecraft have been designed to withstand, with adequate margin, qualification to the static and dynamic Shuttle launch environments.

- Both spacecraft have adequate power to demonstrate a 10% margin, with some left over.
- The RCS thrusters of both spacecraft can meet the 10 mm/s mini-delt-V maneuver requirement provided that the thrusters are match-paired by the supplier before installation on the spaceframe.

A list of conclusions has been appended to the end of the study report, which may be used as an adjunct to this summary.

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Task 1 Candidate Satellite Bus Design

"Identify any candidate designed satellite bus(es) suitable for any or all of the mission options specified in Exhibit 1, entitled Topex Satellite Option Study, dated December 11, 1981."

Two viable bus candidates have been selected for the TOPEX mission, viz.: the P80-1 and the Global Positioning System (GPS) Phase II (operational phase) satellite designs. In the study proposal, Rockwell included a third candidate, the GPS Phase I (validation phase) satellite design; primarily as the candidate for an expendable launch vehicle design. (P80-1 and GPS Phase II are designed for Shuttle Orbiter launch) However, the GPS Phase I design would require major structural and solar array modification to satisfy the requirements of the three mission options listed in Exhibit 1 of the RFP. Therefore, with approval from the JPL TOPEX Project, the GPS Phase I design has been eliminated from further study consideration.

A SUMMARY OF CANDIDATE SPACECRAFT SELECTION DATA

The facing table delineates the attributes and shortcomings of the three Rockwell spacecraft submitted in the study proposal.

CANDIDATE SPACECRAFT SELECTION

CANDIDATES	TCS	EPS	ACDS	PROP	TELCOM/ DATA HND	PAYLOAD ACCOMD.	PROD STATUS
GPS-I	MINOR MOD	MOD SOLAR PANELS LARGER SHUNTS ADDED BATTERY	LARGER WHEELS REPLACE HORIZON SENSOR	OK LARGER RCS TANKS	REPLACE WITH AVAILABLE HARDWARE	20% MODS ADD DEPLOY MECH	IN FINAL ASSEMBLY CHECKOUT PHASE
GPS-II	MINOR MOD	OK	LARGER WHEELS REPLACE HORIZON SENSOR	OK		OK ADD DEPLOY MECH	ASSUMED PRODUCTION GO-AHEAD 28 SHIP SETS JAN 83
P80-1	MINOR MOD	OK	OK	OK		OK	IN FINAL ASSEMBLY CHECKOUT PHASE

- DELETE GPS-I AS CANDIDATE
- RETAIN GPS-II & P80-1 FOR FURTHER EVALUATION

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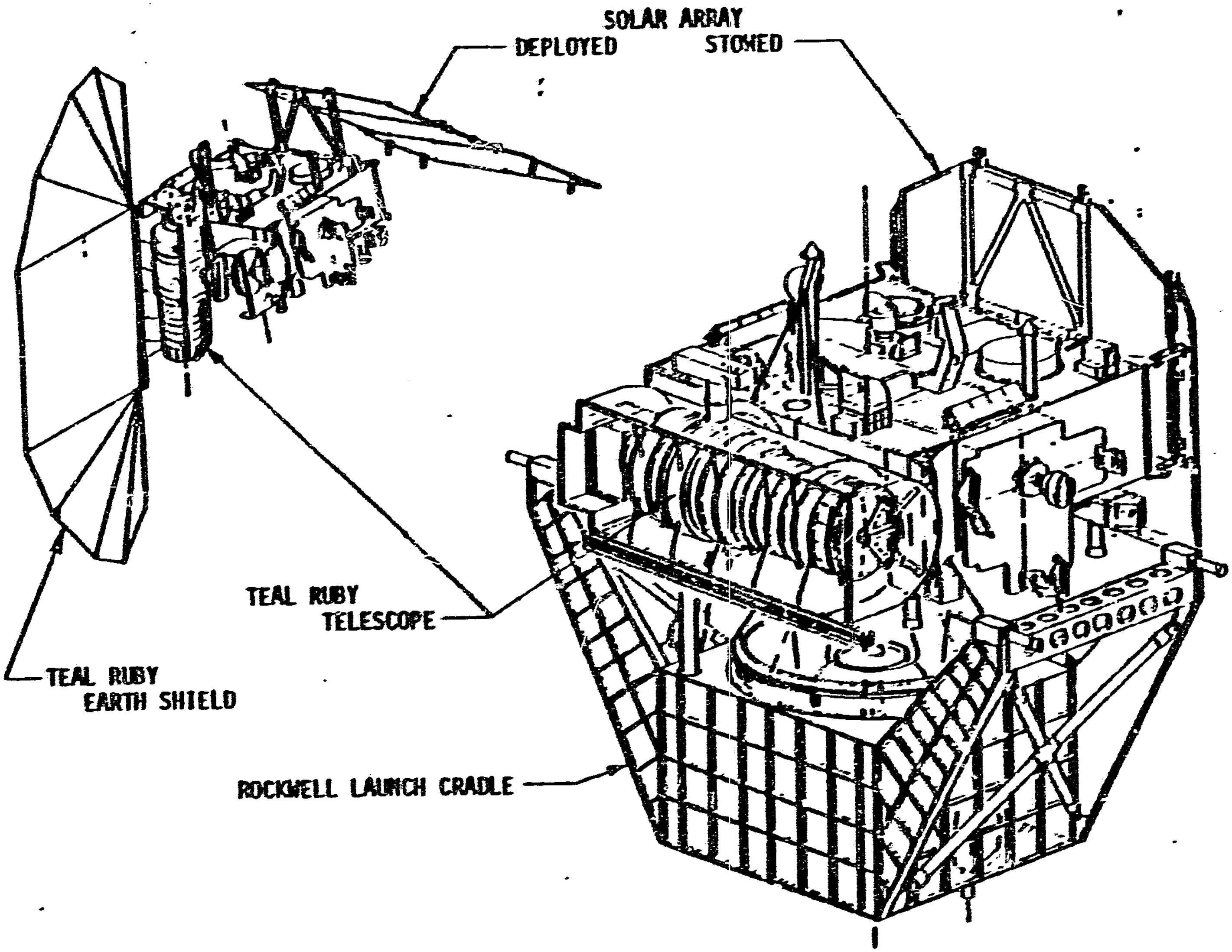
THE AIR FORCE SPACE TEST PROGRAM P80-1 SATELLITE

The P80-1 satellite is shown in the diagram on the facing page. Its prime experiment is the Teal Ruby electro-optical telescope, shown on the -Y side of the spaceframe. Other experiments are also carried and include: a Lasercom Space Measurement Unit (LSMU), Ion Auxiliary Propulsion Unit (IAPS), and an Extreme Ultraviolet Photometer (EUV). The P80-1 is scheduled for launch in the Shuttle Orbiter in August of 1983 and is shown in its launch cradle in a launch-ready condition, with another sketch showing its on-orbit configuration.

The P80-1 was selected because of its payload weight and power capability. Also, its orbit parameters (400 n.mi., with minimum inclination of 72.5°) are close to those required for TOPEX.

THE AIR FORCE SPACE TEST PROGRAM P80-1 SATELLITE



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THE GLOBAL POSITIONING SYSTEM (GPS) OPERATIONAL PHASE SATELLITE

The GPS Phase II satellite is shown in the diagram on the facing page. The prime mission of the GPS Phase II satellite is to generate L-Band RF signals which can be used on the ground, at sea, in the atmosphere and in space for computation of navigation positional data by those equipped with user's receiver/pro processor equipment. A total of 18 satellites will be orbited at an altitude of 10,000 miles (half-synch), six each in three separate orbits 120° in longitude apart and inclined at 55°. Other secondary experiments are also carried which are classified.

The GPS II was selected because of its capability to support the weight and power requirements of the TOPEX mission. Because of its relatively high orbital altitude, some modifications will be required to convert the satellite for low Earth orbit use.

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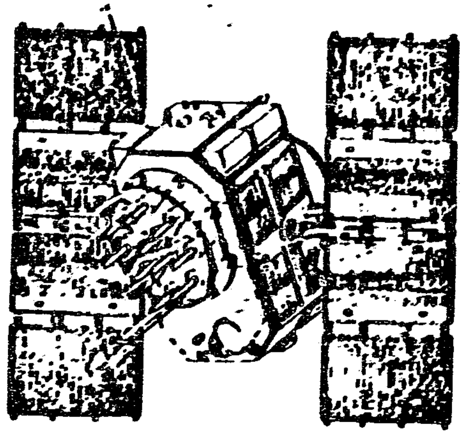
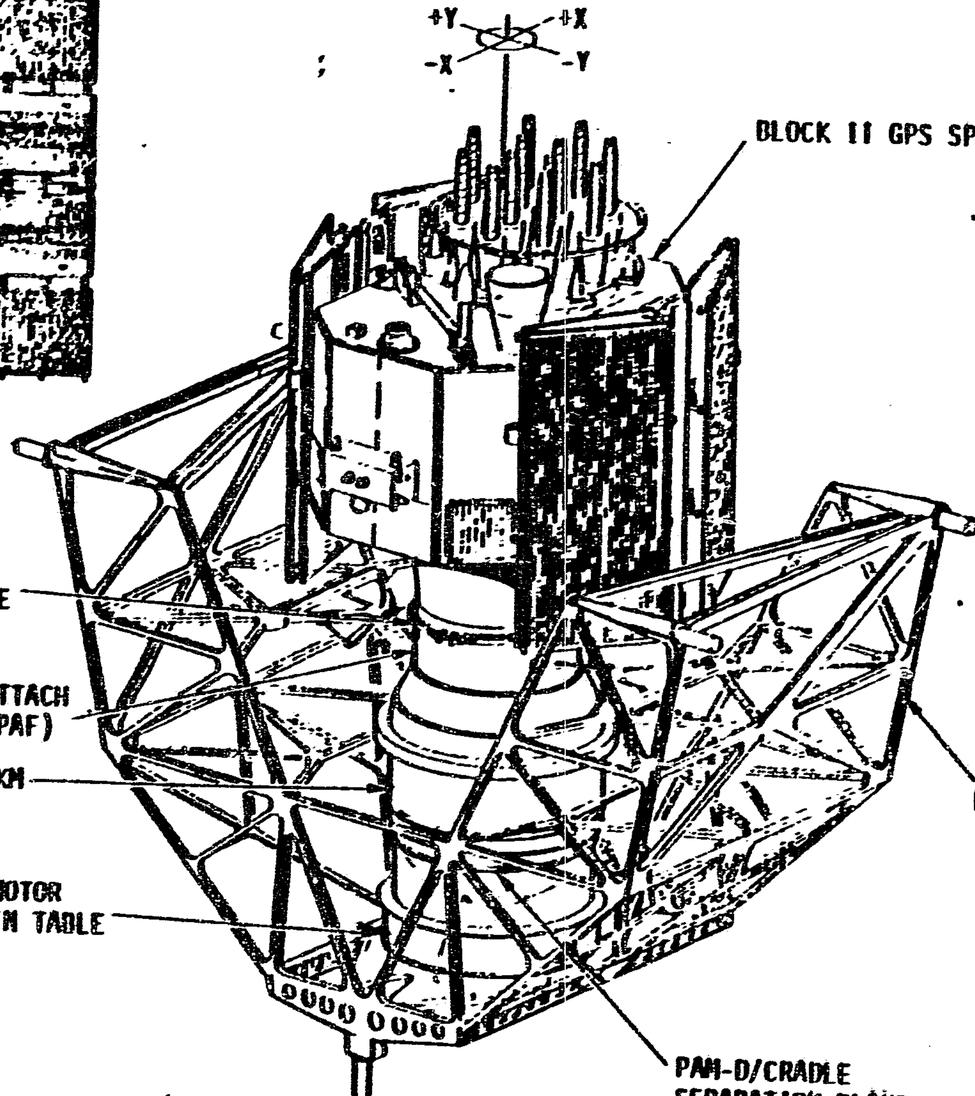
THE GLOBAL POSITIONING SYSTEM (GPS) OPERATIONAL PHASE SATELLITE





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Task 2

"Identify any candidate propulsion Module(s) suitable for any or all of the options specified in Exhibit 1."

The contract satellite requirements specify a Shuttle launch from WSMC (Western Launch Site) to 150 n.mi. at a 63.4° inclination for delivery of the payload to the operational orbits. It is noted that such a condition would be impractical because:

1. Direct insertion of the Shuttle to an inclination of 63.4° is prohibited by Range Safety because of the resultant impact area of the turned out Solid Rocket Motor Boosters and the large External Tank.
2. Insertion into the orbit inclination via an Orbiter plane change to 63.4° would require a propellant load in excess of the Orbital Maneuvering System (OMS) kit installation capacity.

The use of auxillary propulsion, integral with the spacecraft is the most efficient approach.

To accomplish the mission options it will be necessary therefore to provide plane change capability in the Satellite System. As a minimum this is 6.6° out of WTR and 8.4° out of ETR, (for a 70° and 55° launch inclination respectively). The GPS-II bus concept is designed for launch with payloads being delivered to 160 nmi, 28.5° inclination circular orbit.

However, the requirements as stated in the SOW have been discussed. Also, the capability for providing plane change of using the existing Perigee Insertion and Apogee Insertion stages of both P80-1 and GPS Phase II has been addressed. This approach has built-in plane change capability and gives more flexibility to "ride-share" with other payloads to reduce launch costs. It also reduces required modifications to a minimum which reduces the spacecraft development cost. Note that alternative options can be considered to reduce cost. Further analysis is required to determine if alternate options are viable cost savings approaches.

IDENTIFICATION OF PROPULSION MODULES FOR TOPEX

PERIGEE AND APOGEE STAGES FOR P80-1/GPS

Using the given conditions of paragraph III MISSION OPTIONS OF Exhibit 1 of the Statement of Work, the table on the facing page identifies the required propellant weights (without plane change) for all three options for both P80-1 and GPS Phase II spacecraft. The table also identifies the amount of propellant which would be required to be either off-loaded or on-loaded on the identified solid motors.

For this assessment, only the high performance solid propellant rocket motors were considered. It might be possible that a combination of a solid perigee motor and a hydrazine apogee propulsion stage would be more advantageous. This would require larger hydrazine tanks for both the P80-1 and GPS II; solid motors were selected as minimum design change and therefore minimum cost impact.

The candidate motors are manufactured by the Thiokol Corp., and all of them have been qualified in their original configurations.Where on-loading of propellant is indicated, it would require that the motor case be "stretched" to accommodate the additional propellant Where off-loading is indicated, it merely means that the required amount of propellant would have to be removed. Drawings of the three motors are shown. Two each of the same type motor would be used in tandem for perigee and apogee insertion for each mission option.

NEW PERIGEE AND APOGEE PROPULSION MODULES FOR TOPEX MISSION

P80-1 & GPS II

CONDITIONS: SHUTTLE SEPARATION ORBIT CIRCULAR AT 150 N. MI. INCLINED AT 63.4°. ONLY SOLID MOTORS. PROPELLANT WEIGHTS IN PARENTHESES REFER TO THE WEIGHT OF ON-LOADING (+) OR OFF-LOADING (-) REQUIRED.

P80-1

MISSION OPTION	ORBITAL ALTITUDE	PAYLOAD KG (LB)	DELTA-V_A M/S (F/S)	DELTA-V_P M/S (F/S)	S/C WT. ON-ORBIT	PROP. WT. REQ. APOG.	PERIG.	SOLID MOTOR CANDIDATE*
1	1334 KM	190.5 (419.0)	269 (884)	279 (917)	2647 LB	269 LB (+21)	313 LB (+65)	TEM-521-5
2	1000 KM	159.1 (350.0)	192 (629)	197 (645)	2578 LB	182 LB (-66)	203 LB (-45)	TEM-521-5
3	800 KM	209.1 (460.0)	142 (466)	145 (475)	2688 LB	140 LB (13.5)	151 LB (-2.5)	TEM-479

GPS II

SAME AS ABOVE

MISSION OPTION	S/C WT. ON-ORBIT	PROP. WT. REQ. APOG.	PERIG.	SOLID MOTOR CANDIDATE*
1	1889 LB	189 LB (-58.5)	223 LB (-24.5)	TEM-521-5
2	1820 LB	129 LB (-24.5)	144 LB (-9.5)	TEM-479
3	1930 LB	100 LB (+27)	108 LB (+35)	TEM-516

*THIOKOL MOTOR NUMBERS; DRAWINGS OF THESE MOTORS ARE ON THE FOLLOWING CHARTS. BECAUSE OF THE SUPERIORITY OF SOLID MOTORS FOR THESE APPLICATIONS, NO LIQUID STAGES WERE CONSIDERED. ADDITIONAL ORBITAL TRIM WOULD BE SUPPLIED BY THE SPACECRAFT'S REACTION CONTROL SUBSYSTEM.

TOPEX PROPULSION MODULES FOR APOGEE AND PERIGEE INSERTION

P80-1 AND GPS II CANDIDATES

The three solid rocket motors which can satisfy the perigee and apogee insertion requirements for both the P80-1 and GPS II spacecraft are all made by Thiokol, and include the following data:

STAR 17, TEM-479, burn-out (on-orbit) weight = 18.8 lb.

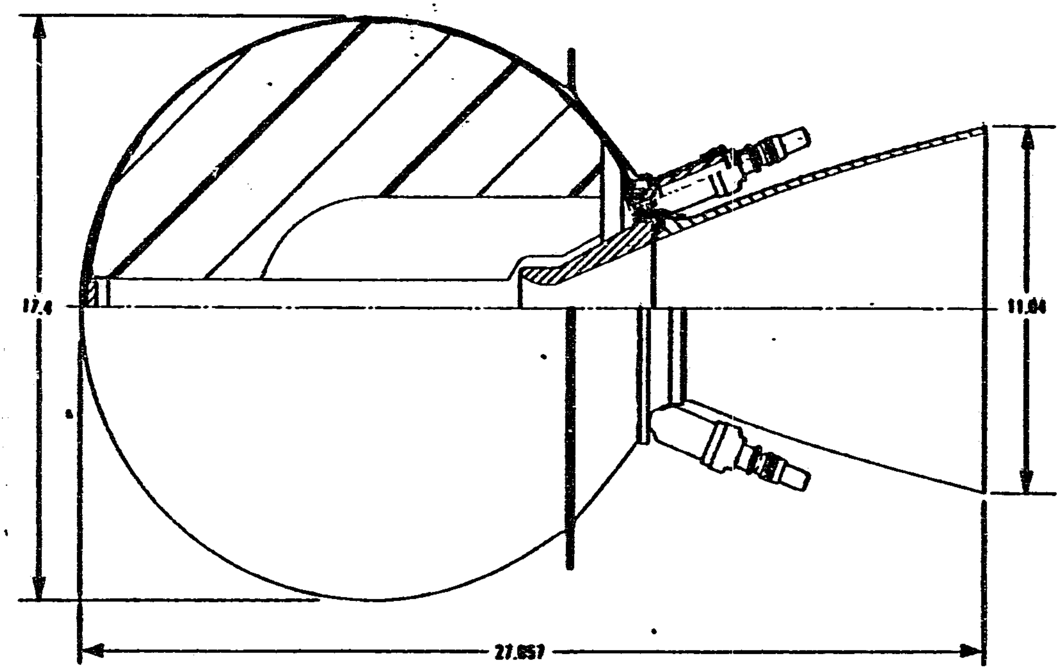
STAR 17A, TEM-521-5, burn-out (on-orbit) weight = 26.5 lb.

STAR 13A, TEM-516, burn-out (on-orbit) weight = 10.0 lb.

Specification drawings of these motors are found on the next three charts.

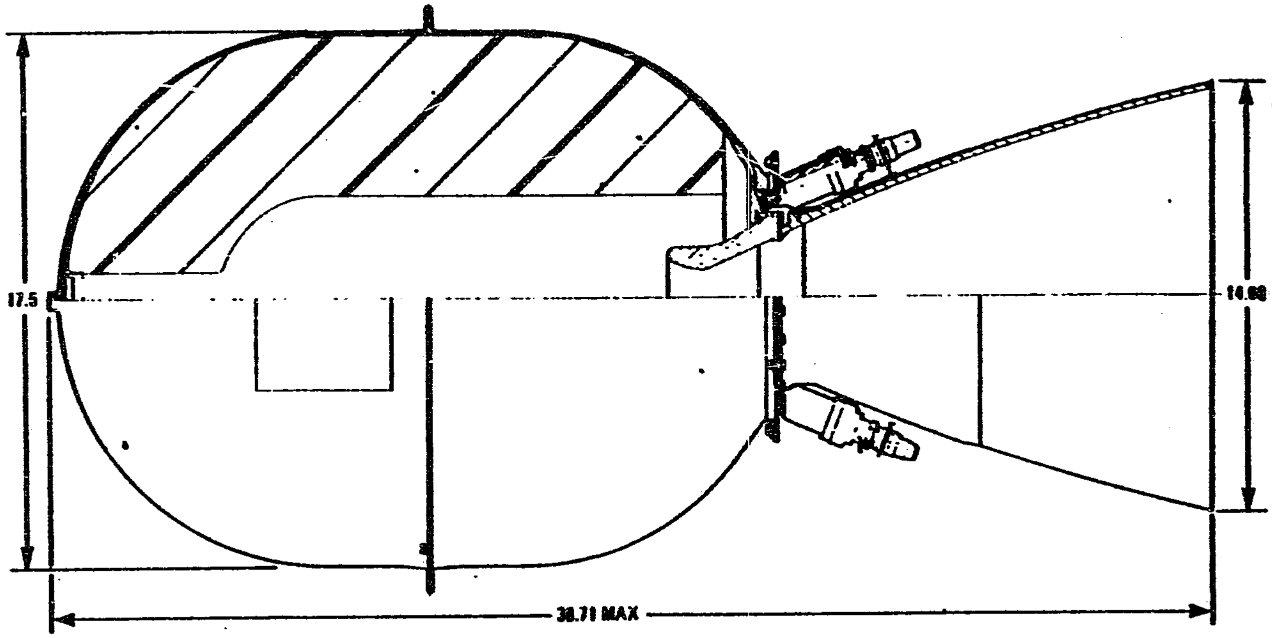
SRM CANDIDATE FOR P80-1 OPTION 3, & GPS OPTION 2

* STAR 17
TE-M-479
17.6-KS-2,460
ORBIT INSERTION MOTOR



SRM CANDIDATE FOR P80-1 OPTIONS 1 & 2, & GPS OPTION 1

* STAR 17A
TE M 521-5
19.4-KS-3,600
APOGEE MOTOR



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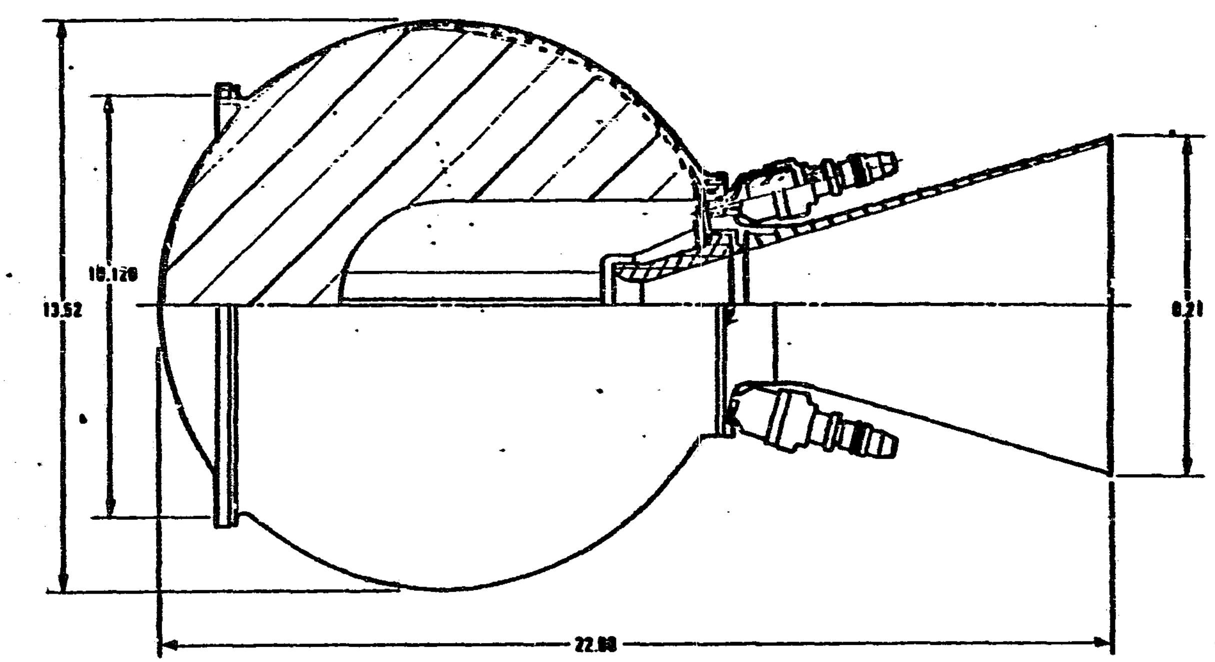
SRM CANDIDATE FOR GPS PHASE II, OPTION 3 ONLY

* STAR 13A
TE-M-516
15.3-KS-1,320
ORBIT INSERTION MOTOR



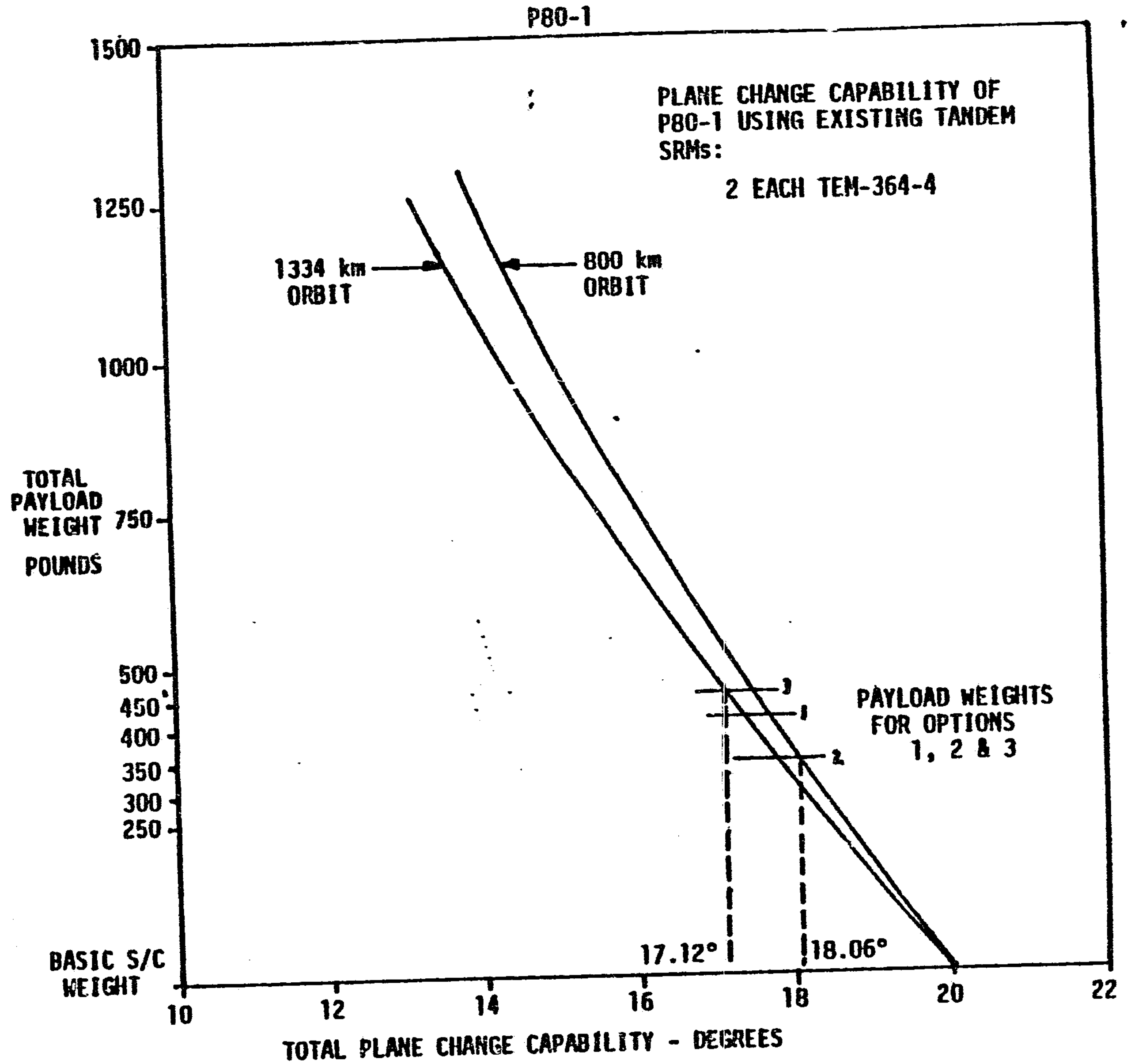
*A PRACTICAL APPROACH FOR PLACING TOPEX IN ITS ORBIT (P80-1)

A direct ascent by Shuttle to the TOPEX orbit of 63.4° is impractical therefore, the following information is presented as a means of providing plane change capability by the satellite to acquire the mission orbit from Shuttle "standard" orbits, but also as a means of "ride-sharing" with other payloads to lower launch costs. If either P80-1 or GPS II perigee and apogee insertion stages were used unchanged, the modification costs would be nil. The plane change capability of P80-1 is shown on the facing chart. The GPS II plane change capability is shown on the chart following this one.

Note that with the heaviest payload option (3) and the highest orbit, P80-1 would contain an excess of delta-V capability to allow a 17.12° plane change. With the lightest payload option (2), lowest orbit, the capability would allow a plane change of 18.06°.

These values are higher than that required to deliver TOPEX from a 55°ETR OR 70°WTR standard Shuttle orbits. However, the solid rocket motors used on either P80-1 or GPS II could be off-loaded and/or non optimum trajectories used to achieve the required capability. Off-loading can be done up to about 25% before the capability of the ignitor would be affected. If more than 25% off-loading is required, ballast or non optimum trajectories incorporated. Smaller motors could also be considered, however, the above option would appear to be lowest cost approach.

ADDITION OF PLANE CHANGE CAPABILITY FOR SHUTTLE 'RIDE SHARING' WITH OTHER PAYLOADS



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A PRACTICAL APPROACH FOR PLACING TOPEX IN ITS ORBIT (GPS-11)

The facing chart shows the plane change capability provided by using the existing GPS II perigee and apogee solid rocket motors. Note that with the heaviest payload lofted to the highest orbit, a plane change of 17.62° exists with the excess capability, and a plane change of 18.37° exists for the lowest weight payload placed in the lowest orbit.

As with P80-1, the two SRMs could be off-loaded up to 25% for use in obtaining the exact plane change capability needed to transfer from a 'standard' orbit to that of 63.4°. If the Shuttle launch to be used to TOPEX is chosen sufficiently early, the motors could be trimmed prior to shipment from the vendor in plenty of time to meet the launch date. As with P80-1, orbit circularization will be done with the Reaction Control Subsystem.

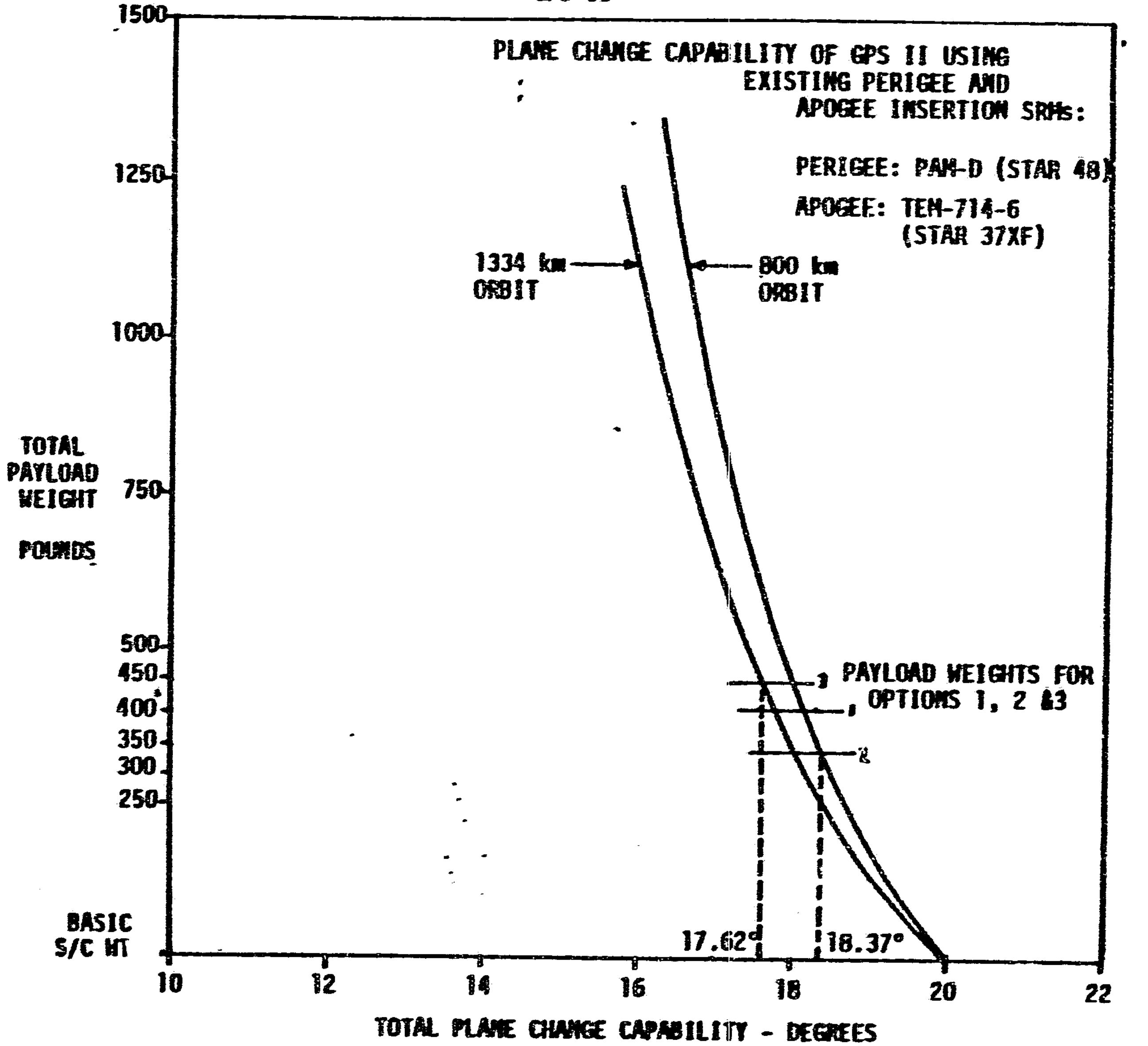
For further study effort, the delta-Vs required for translating from a 'standard' orbit (from either Shuttle launch site) to an inclination of 63.4° at the three different option altitudes are presented in the table below:

DELTA-Vs IN METERS/SEC (FT/SEC)

Shuttle Inclination / TOPEX Orbit Altitude	ELS 28.5°	ELS 55°	WLS 70°
800 KM	ΔV_p 4688.21 (15382.0)	ΔV_p 1153.57 (3784.86)	ΔV_p 911.09 (2989.28)
	ΔV_a 142.02 (465.97)	ΔV_a 142.02 (465.97)	ΔV_a 142.02 (465.97)
1000 KM	ΔV_p 4705.49 (15438.71)	ΔV_p 1164.94 (3822.17)	ΔV_p 923.66 (3030.53)
	ΔV_a 191.66 (628.84)	ΔV_a 191.66 (628.84)	ΔV_a 191.66 (628.84)
1334 KM	ΔV_p 4734.10 (15532.58)	ΔV_p 1187.54 (3896.32)	ΔV_p 949.25 (3114.49)
	ΔV_a 269.35 (883.74)	ΔV_a 269.35 (883.74)	ΔV_a 269.35 (883.74)

ADDITION OF PLANE CHANGE CAPABILITY FOR SHUTTLE 'RIDE SHARING' WITH OTHER PAYLOADS

GPS II



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Task 3 Status of Development

"Indicate the status of development of each candidate; i.e., designed, built, qualified, flown (specify missions)".

The status of the candidate spacecraft for both P80-1 and the GPS Phase II spacecraft are summarized in the following pages. The GPS Phase II data include the heritage/legacy directly derived from the GPS Phase I spacecraft, six of which are presently operational on-orbit.

DEVELOPMENT STATUS OF THE P80-1 SPACECRAFT

The P80-1 spacecraft is one of a series of U.S. Air Force Space Test Program (STP) experimental space vehicle platforms. P80-1 will be the first of the series to be launched in the Shuttle Orbiter. The Space Vehicle (SV) program consists of a single spacecraft (i.e., protoflight qualification) which carries a number of Air Force experiments, the results of which will be applied to future Air Force operational spacecraft.

The facing chart shows the program schedule for P80-1. The Preliminary Design Review (PDR) was held in December of 1978, and the Critical Design Review (CDR) one year later. At present, the P80-1 is on schedule (as shown on the chart), and is in system protoflight qualification testing.

Launch is tentatively scheduled for August of 1983, and the actual launch is subject to Air Force/NASA Shuttle manifest priority agreements. It should be noted that the unusual length of the program resulted from a combination of reduced funding and Shuttle availability. It is projected that a TOPEX version of the P80-1 could easily be accomplished in the "40-month from go-ahead" indicated by a start in August 84 and a launch in November 87.

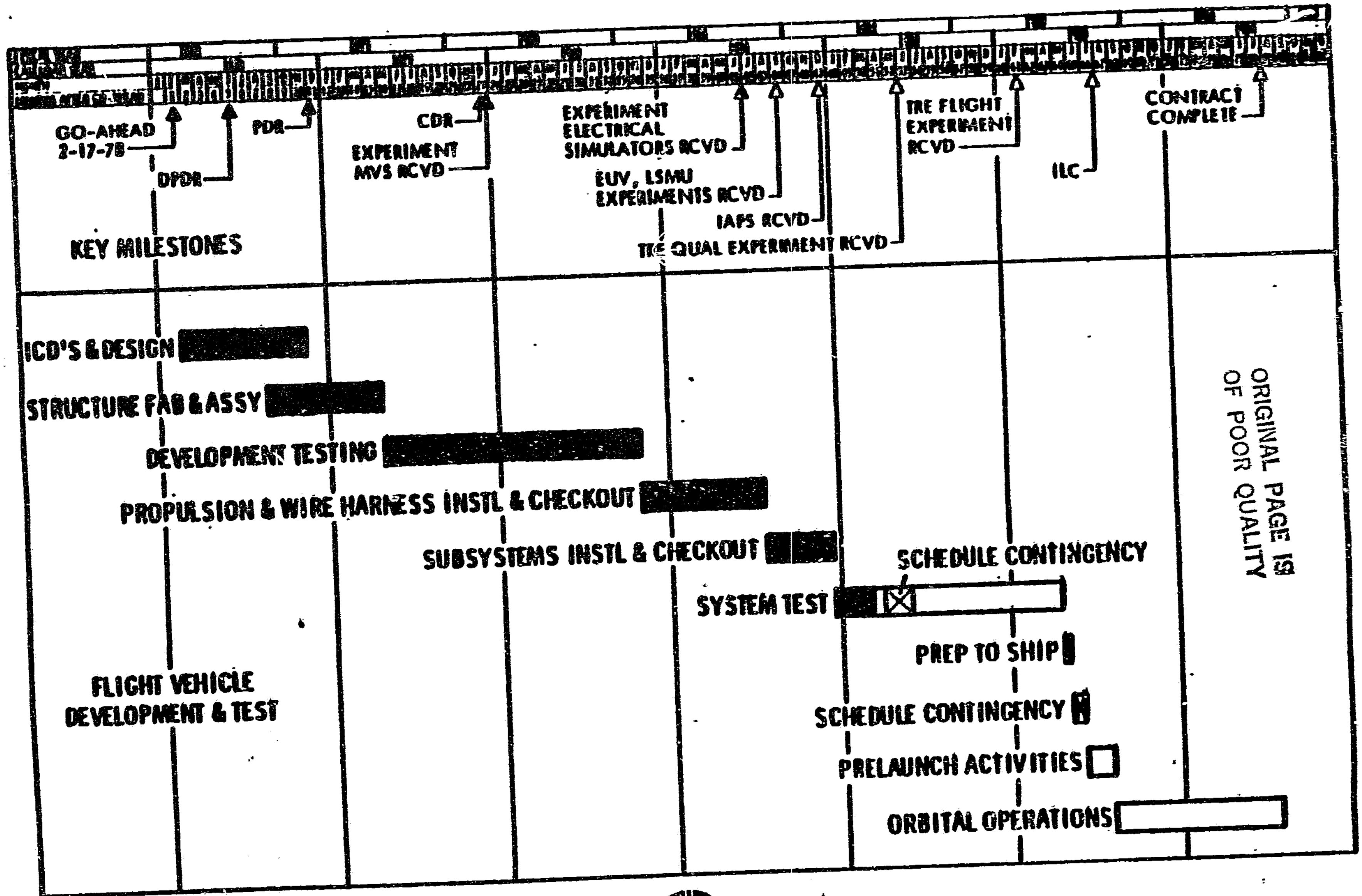

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Figure P80-1 Program Schedule and Development Status

DEVELOPMENT/FLOWN STATUS OF THE GPS NAVSTAR SPACECRAFT

PHASE I & PHASE II

The Phase I GPS NAVSTAR spacecraft comprises the developmental phase of the DoD multi-service Global Positioning System, which are being used to validate the radio frequency navigational concept. Phase II of the program will be the operational phase, which will consist of 18 satellites, three each in six different inclined orbits separated in longitude by 60°. As seen in the facing table, seven Phase I NAVSTARS have been launched since early 1978, with the loss of only one due to the failure of an Atlas booster main engine soon after lift-off. The Phase I Qualification Test Vehicle (QTV) has been refurbished and is now in storage, awaiting a launch as NAVSTAR No. 8 in August of this year (1982). This Flight Space Vehicle (FSV) will be used to test a secondary "piggy back" classified secondary payload for the first time, as well as perform as a navigational vehicle on-orbit. The referenced payload will be included as standard equipment on the Phase II vehicles. Three additional Phase I FSVs are in assembly and checkout.

The first Phase II vehicle, GPS 0012, has been designated as the qualification test vehicle (QTV). As with the Phase I QTV, the Phase II QTV will be qualified to the very stringent MIL-STD-1540A. The severity of this qualification test program is often given credit for the success enjoyed by the Phase I NAVSTARS. The Phase II QTV will repeat this qualification program but with Shuttle environments substituted. The two flights of the Shuttle Orbiter to date, and the planned missions this year, will aid in accurately determining the exact Shuttle launch and orbital environments for the Phase II qualification and acceptance test programs. The Phase II QTV testing will be complete only a few months after the projected TOPEX go-ahead date of 8/1/84.

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GLOBAL POSITIONING SYSTEM PHASE I & PHASE II STATUS

	APPLICATION/NO.	DISPOSITION	DATE	MANU. DESIG.	STATUS
PHASE I	NAVSTAR NO. 1	LAUNCHED	2-22-78	(FSV NO. 1)	IN OPERATION ON-ORBIT
	NAVSTAR NO. 2	LAUNCHED	5-13-78	(FSV NO. 2)	IN OPERATION ON-ORBIT
	NAVSTAR NO. 3	LAUNCHED	10-6-78	(FSV NO. 3)	IN OPERATION ON-ORBIT
	NAVSTAR NO. 4	LAUNCHED	12-10-78	(FSV NO. 4)	IN OPERATION ON-ORBIT
	NAVSTAR NO. 5	LAUNCHED	2-9-80	(FSV NO. 7)	IN OPERATION ON-ORBIT
	NAVSTAR NO. 6	LAUNCHED	4-26-80	(FSV NO. 8)	IN OPERATION ON-ORBIT
	NAVSTAR NO. 7	LAUNCHED	12-19-81	(FSV NO. 5)	BOOSTER FAILURE OFF PAD
	REFURBISHED ØI QUAL. VEHICLE	SCHEDULED LAUNCH AS NAVSTAR 8	8-82	(FSV NO. 6)	IN STORAGE
	GPS 0009, 0010 & 0011	—	—	—	IN PRODUCTION
PHASE II	GPS 0012 (ØII QUAL. VEHICLE)	SCHEDULED QUAL. COMPLETE	LATE CY 1984	—	IN DESIGN
	GPS 0013 - GPS 0040	—	—	—	PLANNED PRODUCTION START FY'83

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Task 4 Performance of Each Candidates

"Describe the performance of each candidate (including performance margins) with respect to requirements specified in Exhibit 1, i.e., payload accommodation, attitude determination and control, telecommunications, command and data handling".

The following subsystems and their performance margins are described for both the P80-1 and GPS Phase II spacecraft in the text, tables and pictorials of this task response. Included are definitions of the required modifications identified in this study; however, it should noted that no effort has been extended to develop design solutions to the required modifications:

- Structure Subsystem
  - Basic structure and payload accommodation
  - System weight statement
- Telecommunications Subsystem (life)
  - Housekeeping and payload telemetry and command
  - TDRSS accommodation
  - Operational orbit determination
- Power Subsystem
  - Payload and subsystems power requirements
  - Battery charge/discharge and eclipse effects
  - Operating voltage requirements
- Reaction Control Subsystem
  - Type and size
  - Operational modes
  - On-orbit expendable capability
- Attitude Control Subsystem
  - Pointing capability
  - TDRSS antenna pointing accommodation
  - Attitude and rate determination
- Thermal Control Subsystem
  - Basic TOPEX thermal approach
  - Payload thermal control

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P80-1 STRUCTURE SUBSYSTEM MODIFICATION FOR TOPEX

The exact dimensions, mounting provisions, layout footprints and other physical data of the science instruments/sensors would have to be known before the detailed modifications to the structure could be determined.

However, it is assumed that the TOPEX payload components could easily be located in places presently occupied by the P80-1 payload items. Thus; little modification would be required for the P80-1 Structure Subsystem to accommodate the TOPEX instrument. Structure weight would change only slightly, so is assumed to be the same as that for the basic P80-1 mission, 729.2 pounds (331.45 Kg), which includes the solar array substrates.

GPS II STRUCTURE SUBSYSTEM MODIFICATION FOR TOPEX

The GPS II Structure Subsystem appears to be compatible with the TOPEX payload components, and therefore minor modifications are anticipated. The only significant problem noted is the placement of the 2-meter radio altimeter antenna (required for Mission Option 1) on the spacecraft payload bulkhead. This diameter of antenna would interfere with the solar array rotation. (No problem exists for the 1-meter antenna required for Mission Options 2 and 3.) If this antenna is selected, then modification to the solar array arm length would be required to provide the necessary clearance.

No design effort was spent during this study to locate the parabolic TDRSS antenna for its steering and view-angle requirements. Some structure modification may be required to do this.

Without further physical description of the payload components, it is assumed that the GPS II structure will not appreciably alter in weight, and will weigh about the same as for the basic GPS II mission, which is 410.6 lb. (186.6 Kg.).

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P80-1 AND GPS II WEIGHT STATEMENTS MODIFIED FOR TOPEX

Only one Mission Option (#3) was chosen for this exercise. Similar weight statements could be made for the TOPEX payload and solid motor apogee and perigee insertion stages.

The weight for ballast was estimated. Solid propellant weights and burn-out weights for the SRMs used for the other two Mission Options (2 & 3) can be found in the table given in Task 2. Alternate propulsion systems are available for providing plane change capability. Further analysis is required to select the best option.

P80-1 and GPS II WEIGHT STATEMENTS MODIFIED FOR TOPEX MISSION

WEIGHT ADDED OR REMOVED (DURING MISSION)	P80-1	GPS II	
TOPEX PAYLOAD (MAX., OPTION 3)	460.0 LB	460.0 LB.	
STRUCTURE	792.2	410.6	
BALLAST	25.0	5.0	(BOTH ESTIMATED)
ELECTRICAL POWER	368.6	289.4	
POWER/SIGNAL WIRE HARNESSES	233.0	123.4	(DOES NOT INCLUDE 68.4 LB. FOR RADIATION HARDNESS)
REACTION CONTROL (DRY)	230.0	51.8	
C&DH/TDRS TELECOMMUNICATIONS	203.5	203.5	
ATTITUDE CONTROL	228.3	87.9	
THERMAL CONTROL	55.0	142.0	
SPACECRAFT (DRY)	2,595.6	1,773.6	
RCS PRESSURANT/PROPELLANT	75.0	93.0	
APOGEE INSERTION SRM EMPTY CASE	---	10.0	(P80-1 JETTISONS ALL MOTOR CASES)
INITIAL ON-ORBIT	2,670.6	1,876.6	
APOGEE INSERTION SRM/PROPELLANT	166.5	100.0	
POST-PERIGEE INSERTION	2,837.1	1,976.6	
PERIGEE INSERTION SRM	166.5	110.0	
SHUTTLE SEPARATION	3,003.6	2,086.6	
SHUTTLE AEROSPACE SUPPORT EQUIPMENT	2,117.0	2,555.0	
CHARGEABLE LIFT-OFF WEIGHT	5,120.6 LB	4,641.6 LB	

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CANDIDATE SPACECRAFT TELECOMMUNICATIONS SUBSYSTEM COMPATIBILITY

& RECOMMENDED TOPEX C&DH/TDRS SUBSYSTEM

Both the P80-1 and GPS II Telemetry, Tracking & Command (TT&C) subsystems were designed specifically for Air Force Space Ground Link System (SGLS) use, and do not lend themselves easily to NASA C&DH/TDRS use. On subsequent pages, some full or partial application of existing TT&C components is discussed, but in general, the TT&C subsystems of either candidate spacecraft will have to be replaced.

The recommended TOPEX C&DH/TDRS subsystem is shown on the facing page. A weight and power table is shown below:

COMPONENT CHARACTERISTICS

ITEM	WT. (LBS)	PWR (W)	UNIT (NO.)
TRANSPONDER	14	12 EA(1) 38.6 TOT	2
RF ASSY	5	—	1
OMNI ANT. SYS.	1.5	—	1
PARABOLIC ANT.	20(2)	—	1
C&DH PROC.	29	52	1
REMOTE D.A.U.	4	10	7
TIMING UNIT	6	12	1
MASS MEM CTL	8	12	1
EMI FILTER	4	—	1
TAPE RECORDER	22	30 REC	2
TAPE ELECT. UNIT	15	45 P/B	2
TOTAL	203.5	208.6	21

(1) 5 WATTS RF, RCVR ON

(2) INCLUDES GIMBAL DRIVE

The narrow-band module contains the RF assembly as well as the transponders. The C&DH Processor shown interfaces with the instruments and subsystems via a data bus and the Remote Data Acquisition Units. Mass Memory and Data Storage devices are used for telemetry and commands.

The data bus concept will eliminate many pounds in harness weight vs. the hardwired point-to-point design of the existing P80-1 and GPS II TT&C subsystems.

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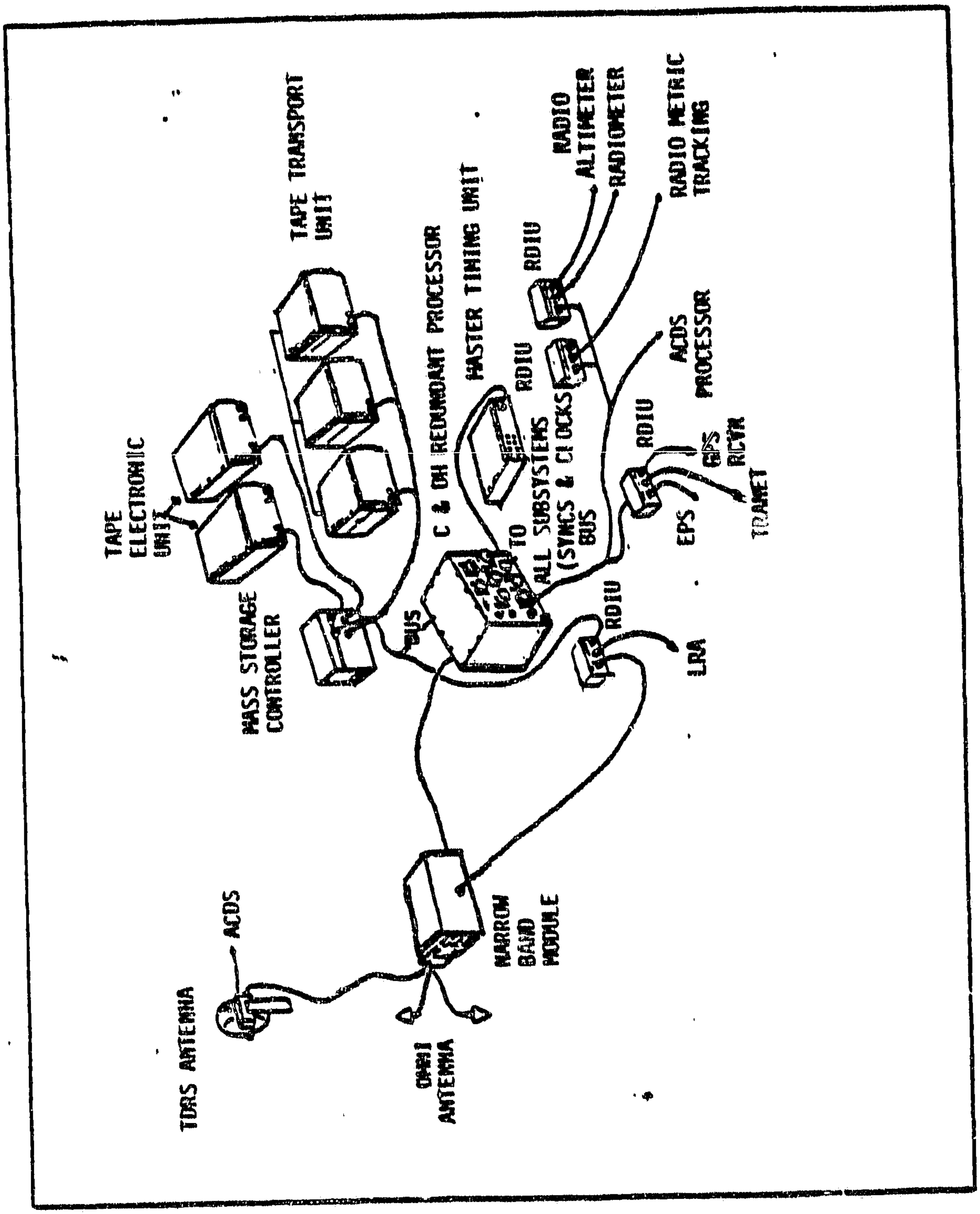


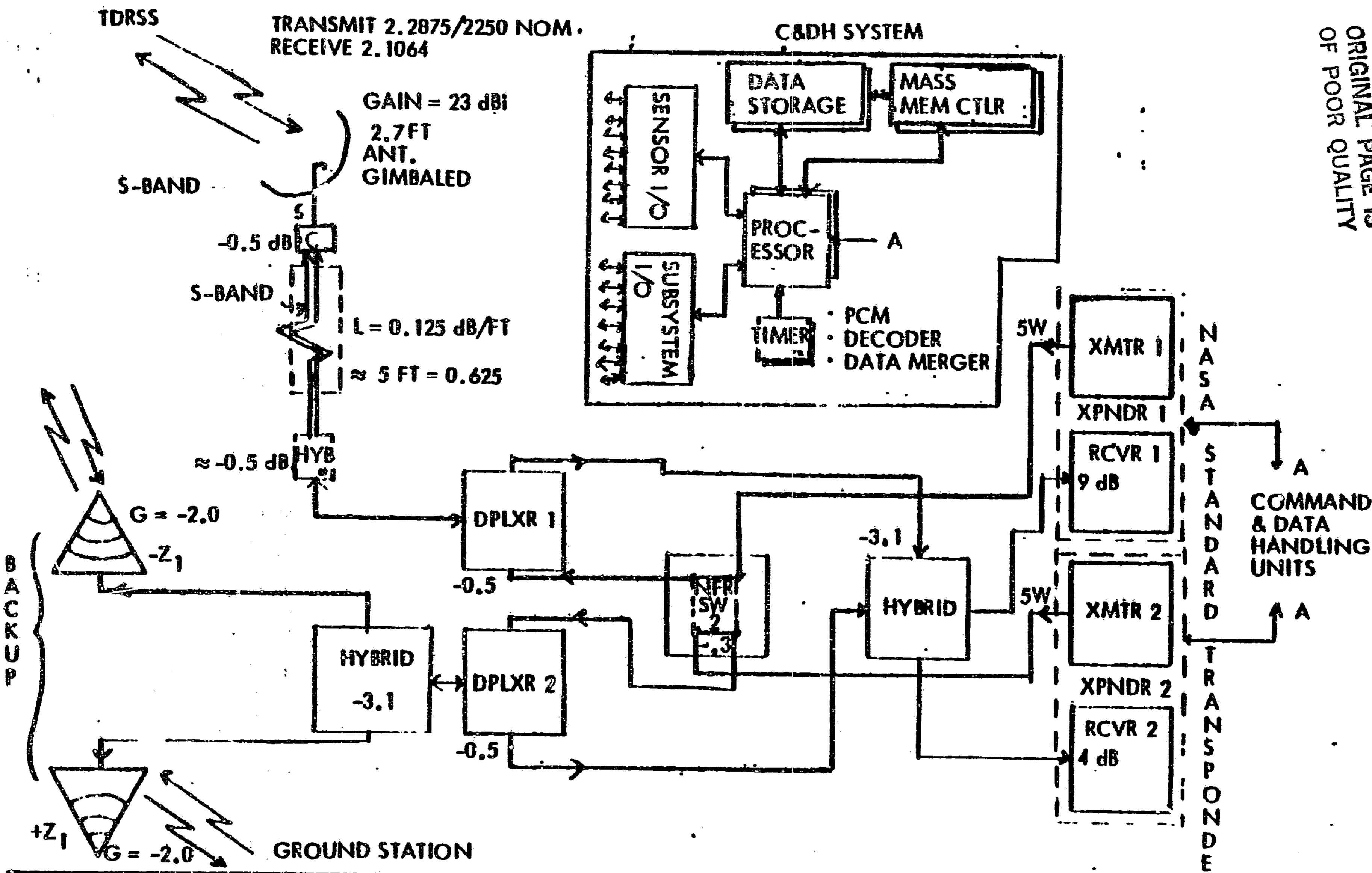
CANDIDATE C&DH/TDRS SUBSYSTEM FOR P80-1/GPS II TOPEX

TOPEX C&DH/TDRS SUBSYSTEM BLOCK DIAGRAM

The C&DH/TDRS subsystem RF and data processing functions are shown in the diagram on the facing page, and illustrate the basic elements with the associated characteristics, (i.e., output power, insertion loss, gain, etc.), of a baseline subsystem. Redundancy is also shown for the transponders and other active elements.

Either the high-gain or omin-antennas may be excited by the transmitters and both receivers are always connected to the antenna system. Redundancy is also shown in the cable to the parabolic antenna, as it must be positioned on a mast or other device.

TOPEX C&DH/TELECOMMUNICATIONS SYSTEMS BLOCK DIAGRAM



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COMPARISON OF TELECOMMUNICATIONS CHARACTERISTICS

TDRSS/STDN : SGLS

The prime difference between the GPS/P80-1 telecommunications characteristics and requirements is the ground system compatibility. Both candidate spacecraft were designed for use by the Air Force Satellite Control Facility (AFSCF) via the Space Ground Link System (SGLS). TOPEX will use the TDRSS relay satellite, with the Goddard STDN as a back-up.

The two TT&C subsystems also employe encryption devices for downlinked data, and decryption devices for uplinked commands. Such security measures are not required for the unclassified TOPEX mission, and will not be included. The TT&C transponders, although they operate in the same frequency band, would not be compatible with the NASA C&DH/TDRS system.

The facing page shows a table comparing the TDRSS, GSTDN and AFSCF-SGLS ground systems.

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TDRSS/STDN-SGLS; COMPARISON OF C&DH/TELECOMMUNICATIONS CHARACTERISTICS

PARAMETER	TDRSS	GSTDN	SCF-SGLS
UPLINK FREQUENCY (FORWARD)	MA 2106.4; SSA 2025-2120 MHz KSA 13.775 GHz	2025-2520 MHz	1763.7-1839.8 MHz
RF POWER (U/L, F/L)	MA: ADR = 25.5 + G,T_S = 824°K SSA: ADR = 35.1 + G,T_S = 824°K KSA: ADR = 20.7 + G,T_S = 893°K @ BER = 10^{-5}, 3 dB MARGIN	10 KW	10 KW
UPLINK MODULATION	QPSK, PN CODE	PHASE MODULATION + S/C	TONE PHASE MODULATION + PRN
UPLINK POLARIZATION	MA: LCP; SSA/KSA: RCP, LCP	R & LHCP	LHCP
DOWNLINK TO UPLINK RATIO	240/221	240/221	256/205
DOWNLINK FREQUENCY (RETURN)	MA: 2287.5 SSA: 2200-2300 KSA: 15.0 GHz	2200-2300	2200-2300
RECEIVE G/T	MA: ADR = EIRP + 35.7 SSA: ADR = EIRP + 24.6 KSA: ADR = EIRP + 30.3	9 MTR: 24.2	9 MTR: 24.1
DOWNLINK MODULATION	QPSK/SQPK	PHASE MOD CARRIER SUBCARRIER BI-ϕ MOD	PHASE MOD SUBCARRIERS ON CARRIER; SUBCARRIER BI-ϕ OR FM MODULATED
CMD DATA TYPE CMD DATA RATE	NRZ MA: 10 KBPS; SSA 300 KBPS; KSA: 2.5 MBPS	NRZ $\leq$ 2 KBPS	NRZ $\leq$ 2 KBPS
TLM DATA TYPE TLM DATA RATE	NRZ & Bi-ϕ MA: 50 KBPS; SSA: 12 MBPS; KSA: 300 MBPS	NRZ & Bi-ϕ 5 MBPS	RZ, NRZ & BI-ϕ 128 KBPS, 256 KBPS, 1024 KBPS

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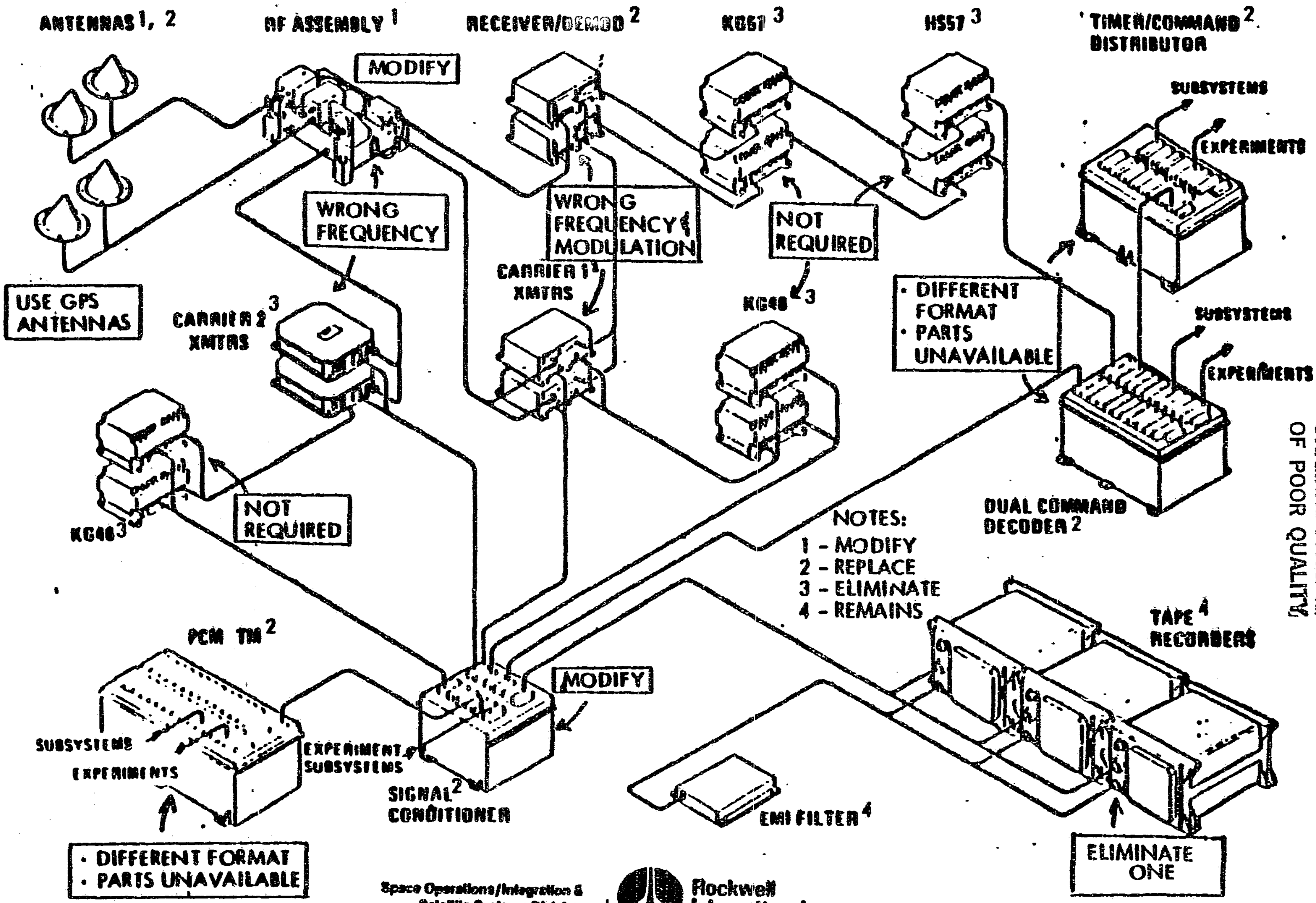
P80-1 TT&C HARDWARE COMPATIBILITY FOR TOPEX

The diagram on the facing page illustrates the basic P80-1 Telemetry, Tracking and Command (TT&C) subsystem, and the various modifications, eliminations, and use-as-is considerations. As shown, most of the P80-1 TT&C components would have no or little application for the TOPEX C&DH/TDRS.

Those components which can be used across-the-board are two of the three tape recorders and their EMI filter.

P80-TOPEX C&DH HARDWARE COMPATIBILITY



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GPS II TT&C HARDWARE COMPATIBILITY FOR TOPEX

The hardware components of the GPS II TT&C subsystem are shown on the facing diagram. Those components that are compatible, modified or deleted are identified. The RF assembly and omni-antenna system may be compatible as-is (or with slight modification).

Encryption/decryption devices will be deleted, as will the SGLS transponders and signal conditioner (which performs GPS II unique signal buffering).

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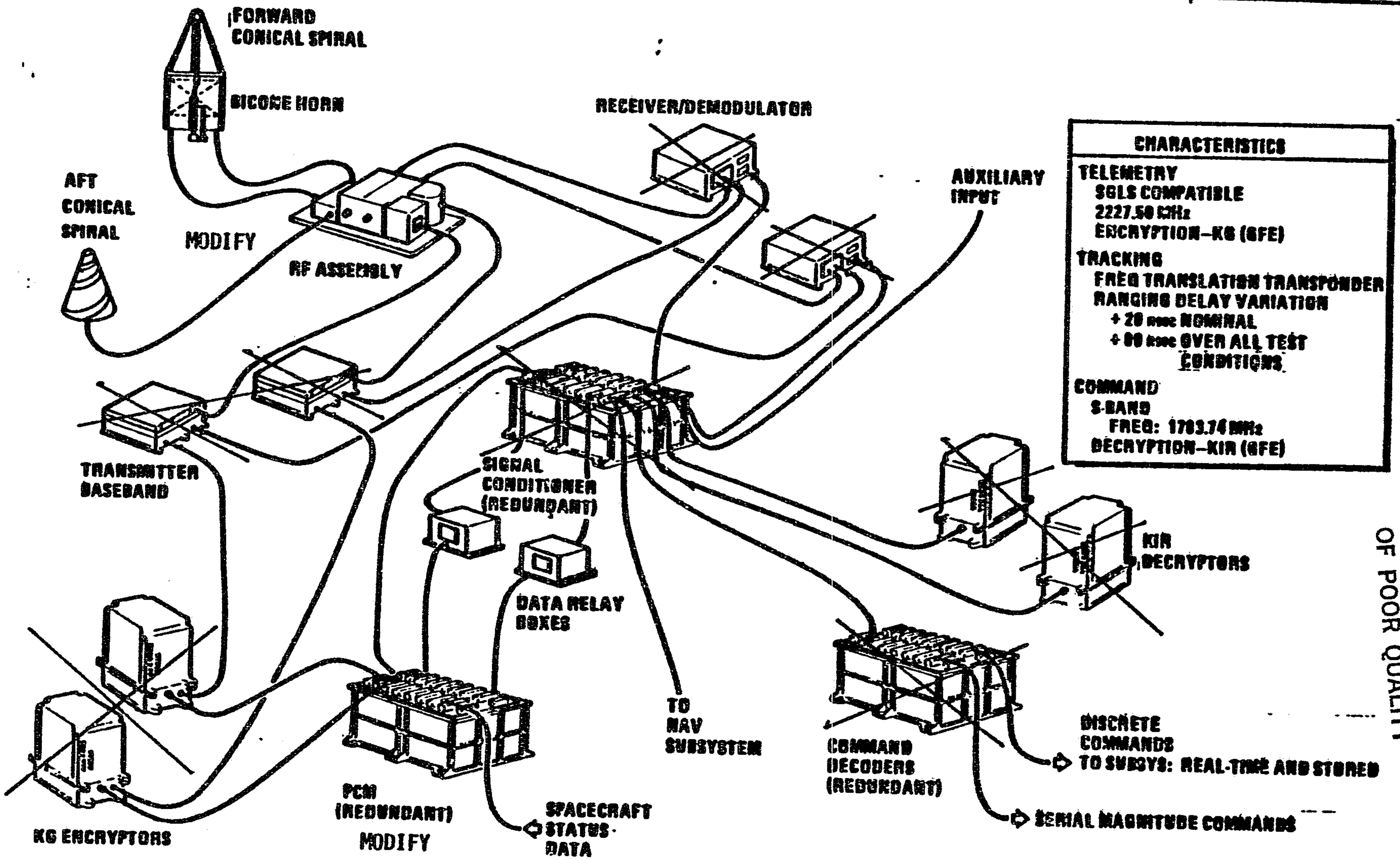
TELEMETRY, TRACKING & COMMAND (TT&C) SUBSYSTEM

GPS II



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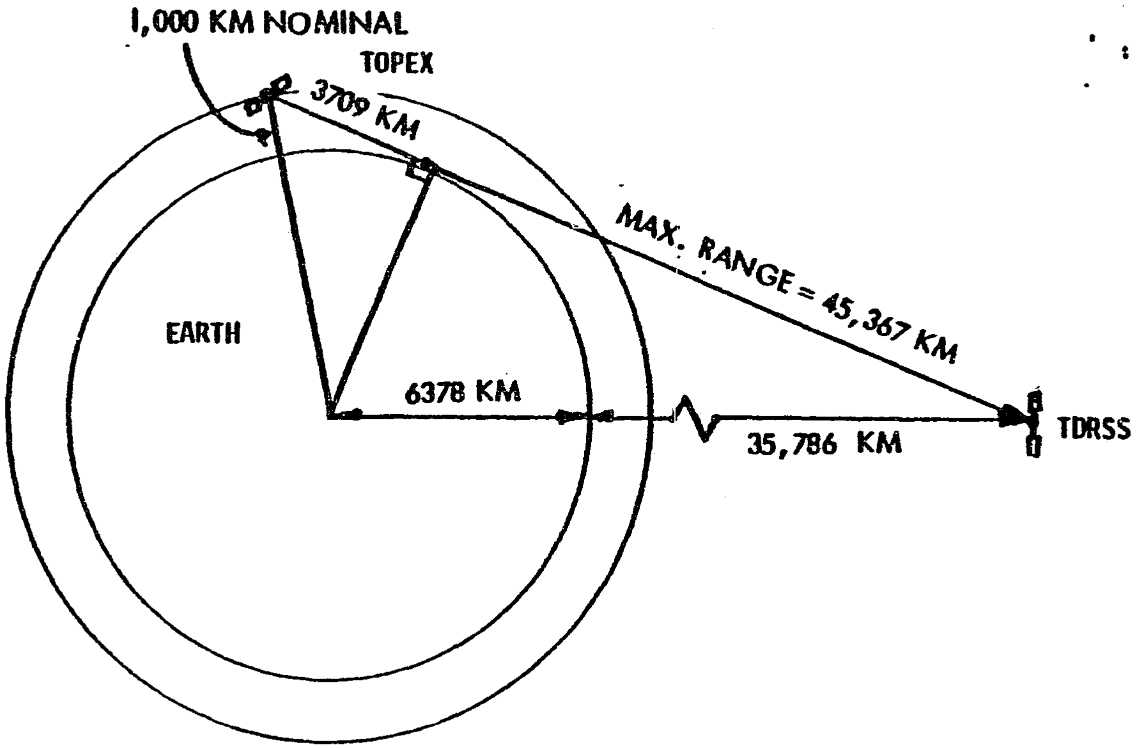
TOPEX - TDRSS/GSTDN RANGE ANALYSIS

The maximum worst-case range to both the TDRSS satellite and a STDN ground station is calculated for a nominal 1,000 Km orbit altitude for the TOPEX satellite.

These ranges are used to calculate link margins to determine transponder requirements.

TOPEX - TDRSS/GSTDN RANGE ANALYSIS



TOPEX - TDRS RANGE = 45,367 KM MAX.

TOPEX - GSTDN SLANT RANGE = 3,709 KM MAX.

TOPEX - TDRSS/GSTDN COMMAND AND TELEMETRY LINK MARGIN ANALYSES

Sample link margins to both the ground system (GSTDN) and TDRSS satellite for command and telemetry data is shown in the following three tables. TOPEX to TDRSS link margins are shown in the facing table, while command and telemetry link margins are covered in the following two tables.

The link margins generate telecommunications system requirements, (i.e., antenna gain, transmitter output, receiver noise temperature, etc.). Both omni and high-gain antenna systems are included with high and low data rate modes.

Design points used are 5 watt transmitters with -2dBi omni and +23 dBi parabolic or helical array.

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TOPEX-TDRSS LINK MARGINS

PARAMETER	UNITS	MODES: OMNI-L/R DATA	PARABOLIC-L/R	PARABOLIC-H/R	REMARKS
		SSA	MA	SSA	
FREQUENCY	MHz	2287.5	2287.5	2250	
DATA RATE	dB-Hz	33.0	46.8	56.8	2 KBPS; 48 KBPS; 480 KBPS
STDN EIRP*	dBw	-2.7	+22.2	+21.1	SSA = ADR - 35.7 MA = ADR - 24.6
FREE SPACE LOSS	dB ( )	0. 6	0.6	0.6	RANGE 45,367 KM MAX.
POL. LOSS	dB	0.5	0.5	0.5	NOMINAL
POINTING LOSS	dB	0.5	0.5	0.5	NOMINAL
MARGIN	dB	3.0	3.0	3.0	
REQD. EIRP	dBw	+1.9	+26.8	25.7	
XMTR POWER	dBw	+7.0	+7.0	+7.0	5 WATTS
CIRCUIT LOSS	dB	-1.1	-2.5	-2.5	
ANT. GAIN	dBi	-4.0	+23.1	+23.1	PARABOLIC 2.7 FT
EIRP	dBw	+1.9	26.6	26.6	3 dB BEAMWIDTH = 11.2^0
ABOVE MARGIN		+0.0	+0.0	+1.8	
6 KBPS	dB-Hz	37.78			
MARGIN	dB	-1.58			

* BER = 10^{-5}

TOPEX COMMAND LINK ANALYSIS

PARAMETER	UNITS	TDRSS VIA OMNI	TDRSS VIA PARABOLIC	GSTDN VIA OMNI	REMARKS
		SSA	MA		
XMTR OUT	dBw	INC.	INC.	40.0	10 KW
CIRCUIT LOSS	dB	INC.	INC.	INC.	
XMTR ANT. GAIN	dBi	INC.	INC.	43.33	9 MTR
EIRP	dBw	+43.6 (45?)	+34.0	+83.2	
SPACE LOSS	dB	-192.6	-192.6	-170	45,367 KM, 3,709KM
ATM. LOSS	dB	—	—	-0.5	NOMINAL
POL. LOSS	dB	-0.5	-0.5	-0.5	NOMINAL
POINTING LOSS	dB	-0.5	-0.5	-0.5	NOMINAL
ANT. GAIN	dBi	-2.0	+22.6	-2.0	
CIRCUIT LOSS	dB	06.7	-5.5	-6.7	
RCVR INPUT	dBw	-158.7	-142.5	-94	
RECEIVE SYST.	dB-^{0}K	+28	+28	+28	
DATA RATE	dB-Hz	+30	+30	+30	KBPS
BOLTZMANNS	dBw	-228.6	-228.6	-228.6	
NOISE POWER	dBw	-170.6	-170.6	-170.6	
E_b/N_0	dB	+11.9	+28.1	+73.6	
HW LOSS	dB	-3.7	-3.7	-3.7	
REQD E_b/N_0	dB	-9.9	-9.9	-9.9	
MARGIN	dB	-1.7	+14.5	+60	

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TOPEX-GROUND TELEMETRY LINK ANALYSIS

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PARAMETER	UNITS	TOPEX-GSTDN	REMARKS
FREQUENCY	MHz	2250	
TRANSMITTER OUTPUT	dBw	7.0	5 WATTS
CIRCUIT LOSS	dB	-4.0	
ANTENNA GAIN	dBI	-2.0	
EIRP		+1.0	
FREE SPACE LOSS	dB	-170.8	
ATMOSPHERIC LOSS	dB	-0.5	NOMINAL
POLARIZATION LOSS	dB	-0.5	NOMINAL
POINTING LOSS	dB	-0.5	NOMINAL
TOTAL XMISSION LOSS	dB	-172.3	
RECEIVE ANT GAIN	dBI	44.1	9 MTR
CIRCUIT LOSS	dB	INCL.	
RECEIVER INPUT	dBw	-125.2	GSTDN
RECEIVER SYSTEM	dB-^{0}K	23.2	
BOLTZMANNS CONSTANT	dBw	-228.6	
DATA RATE	dB-Hz	56.81	480 KBPS
NOISE POWER	dBw	-148.59	
E_b/N	dB	20.38	
E_b/N REQUIRED	dB	9.4	BER = 10^{-5}
MARGIN	dB	+10.98	

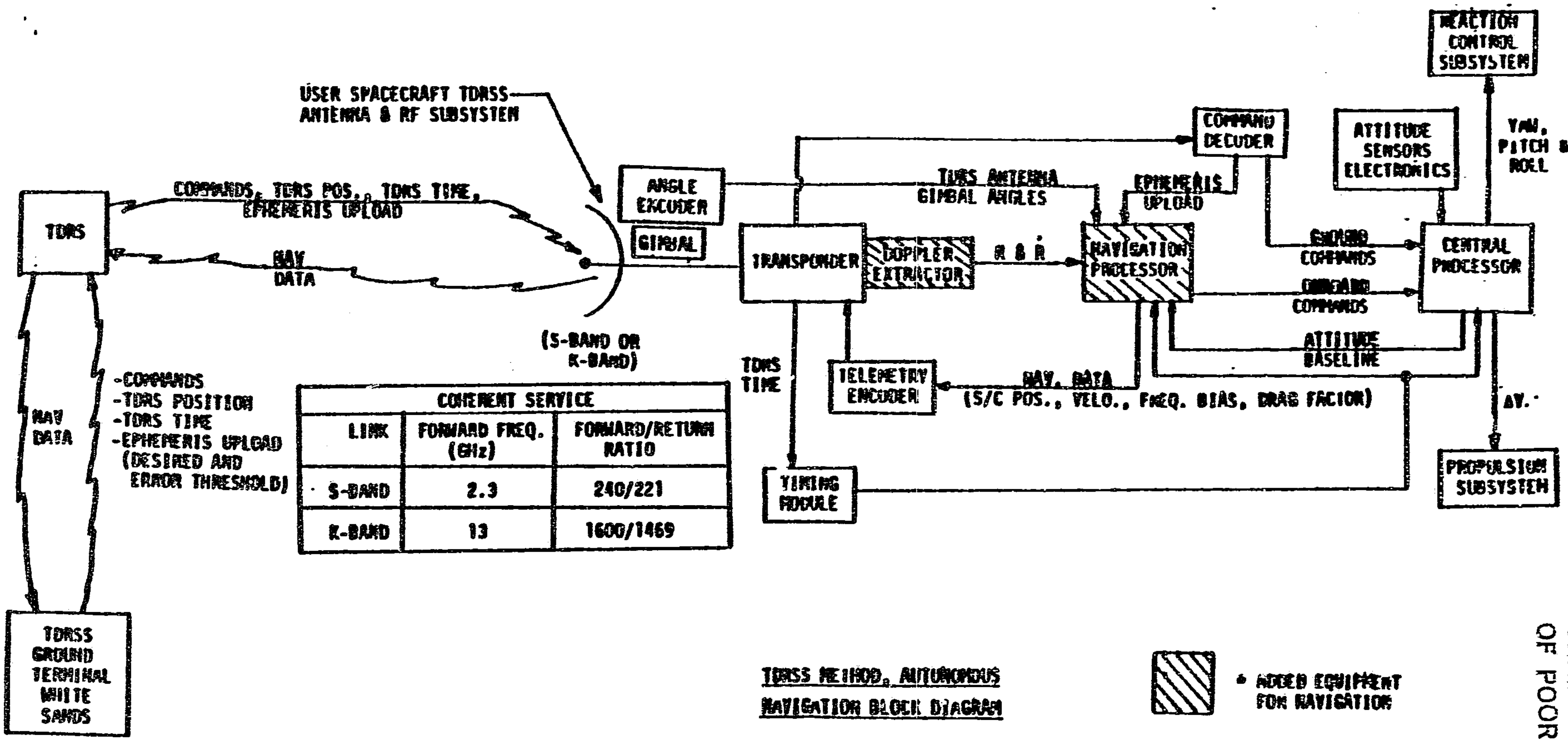
TOPEX - SPACEFLIGHT NAVIGATION USING TDRSS RF DATA

TDRSS RF data may be used to determine spacecraft instantaneous position such that without or with little help from the ground, performance of orbit circularization, orbit plane and alititude change and normal stationkeeping may be accomplished.

Using TDRSS ground angle tracking and coherent RF signals to/from TDRSS, range and range rate is determined and TDRSS orbital position is relayed to the TOPEX satellite. As shown on the diagram on the facing page, a Doppler extractor is required to be added to the C&DH/TDRS subsystem, as well as navigation processor. The angle encoder on the TOPEX gimballed TDRS antenna reveals the inertial space vector between the TOPEX satellite and the TDRSS. Using the TOPEX on-board Attitude Control subsystem pointing information as a baseline attitude to bias the angle encoder, the TOPEX navigation processor can then calculate the TOPEX latitude, longitude and altitude. By repeating the measurements and integrating the data over a period of time, using bith TDRSS satellites, accuracy under 30 meters may be achieved.

The resultant navigation data may be used on-board or relayed to ground control for correction maneuvers.

TOPEX OPERATIONAL ORBIT DETERMINATION USING TDRSS



COHERENT SERVICE		
LINK	FORWARD FREQ. (GHz)	FORWARD/RETURN RATIO
S-BAND	2.3	240/221
K-BAND	13	1600/1469

TDRSS METHOD, AUTONOMOUS NAVIGATION BLOCK DIAGRAM

[hatched box] = ADDED EQUIPMENT FOR NAVIGATION

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ELECTRICAL POWER SUBSYSTEM - TOPEX MISSION ASSUMPTIONS

The following assumptions were used in determining the application/modifications for the P80-1 and GPS II Electrical Power Subsystems (EPS) to conduct the TOPEX mission:

- Eclipse operation of the TOPEX payload instruments is required
- No data need be down-linked during eclipse
- The TOPEX payload demand values given in Exhibit 1 include any necessary power margins
- The maximum orbital period of 112 minutes, with an eclipse period of 34.7 minutes max.
- Downlink communication periods are 22 minutes long
- The C&DH/TDRS subsystem draws a maximum of 162.1 watts when downlinking (no recording during downlink), and 137 watts in eclipse
- Proving capability to meet the maximum payload power requirement (259 watts, Option 1) will satisfy the requirements of the other two options

The detailed capabilities of the P80-1 and GPS II EPS systems are contained in Task 5, but are outlined on the facing page for convenience.

BASIC FEATURES AND CAPABILITIES OF THE P80-1 & GPS II EPS

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P80-1

- DESIGNED TO OPERATE AT HIGH ORBIT INCLINATIONS
- 24 to 33 VDC AT LOADS*
- SOLAR ARRAY - SINGLE DEGREE OF FREEDOM, 35^{o} TILT TO ROTATION AXIS
- TWO 35-AH, 22 CELL Ni-Cd BATTERIES
- DIRECT ENERGY TRANSFER FROM SOLAR ARRAY AND BATTERIES
- FULL SEQUENCED SOLAR ARRAY SHUNT (COMMANDABLE, TEMPERATURE-COMP.)
- LOAD CONTROL - CENTRAL ON/OFF SWITCHING, MAIN BUS LOAD FAULT ISOLATION, AUTO LOAD SHED
- PYROTECHNIC CONTROLLER - CENTRAL ARM/FIRE COMMAND

*MAY REQUIRE MODIFICATION TO MEET PAYLOAD RANGE OF 24 - 32 VDC.

GPS II

- REGULATED 27.0 ± 1.0 VDC MAIN BUS
- 3 35-AH Ni-Cd BATTERIES
- 900 WATT EOL SOLAR ARRAY**
- ON/OFF LOAD CONTROL WITH LOAD SHED OF NON-CRITICAL LOADS
- MAIN BUS LOAD FAULT CONTROL
- SAFE/ARM/FIRE PYROTECHNIC CONTROL
- SELECTED LOAD CURRENT MONITORING
- 3 DEDICATED BATTERY CHARGERS
- 3 BATTERY BOOST CONVERTERS
- AUTOMATIC CHARGE/DISCHARGE CONTROL
- LOAD SHED DETECTION

** WITH THE ADDITION OF MORE CELLS TO FILL EMPTY AREAS ON EXISTING SOLAR SUBSTRATES

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P80-1 POWER REQUIREMENTS SUMMARY, TOPEX

The difference between array demand and EOL output leaves a margin of only 42.5 watts, but there are a number of steps that can be taken to icrease that margin:

- o Eliminate the unnecessary components in the Attitude Control & Determination Subsystem, (see ACDS write-up);
- o Use latest solar cells, and design string layout for maximum output;
- o Duty-cycle all or some of the payload instruments during eclipse;
- o Take no payload data during eclipse.

The last two are not recommended. The most feasible is the ACDS fix; the easiest, most cost effective method to reduce the power load.

The EOL of 3 years also means that to take the 5 year option would require increasing the solar array area for P80-1. This wouldn't be too difficult, but would involve additional cost.

The batteries are more than enough to fill mission requirements. Even if one battery failed, the other battery could support the mission easily with a 28% d-o-d over the 19,000 eclipse periods of the mission. A problem on one battery, however, would be its reconditioning. This could be handled by giving up data-taking during eclipse until the reconditioning was complete.

The IAPS Boost Voltage Converter (see diagram in Task 5) would not be needed for the TOPEX mission and could be eliminated.

Because the P80-1 EPS is a Direct Energy Transfer design, its normal operating voltage range is 24 to 33 VDC. The TOPEX experiments require a range no greater than 24-32 VDC. Thus, a slight modification will be required to bring the P80-1 EPS range into line with that required by the payload.

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P80-1 POWER REQUIREMENTS SUMMARY FOR TOPEX

SUBSYSTEM/LOAD	DATA DOWNLINK (SUNLIGHT)	NO DOWNLINK (ECLIPSE)
C&DH/TDRS	154.9 WATTS (AVE.)*	137.0 WATTS
ATTITUDE CONTROL	134.0	134.0
ELECTRICAL POWER	17.9	17.9
THERMAL CONTROL	10.0 (AVE)	10.0 (AVE)
SUB-TOTAL	316.8	298.9
10% POWER MARGIN	31.7	--
TOPEX PAYLOAD	259.0	259.0
TOTAL (P/L + SUBS. + MARG.)	607.5 WATTS	557.9 WATTS

BATTERY DRAIN DURING ECLIPSE = 0.58 HRS (34.7 MINS) X 557.9 WATTS = 323.6 WATT-HOURS

BATTERY CHARGING POWER NEEDED FOR MINIMUM TIME IN SUNLIGHT = 323.6 WH ÷ 1.29 HRS (77.3 MINS) = 250.8 WATTS ALLOWING FOR ENERGY TRANSFER AND CHARGING INEFFICIENCY, 310 WATTS WOULD BE NEEDED TO RECHARGE THE BATTERIES IN 77.3 M' . ADDED TO THE PAYLOAD AND SUBSYSTEM TOTAL, THE ARRAY DEMAND IS 917.5 WATTS. NOMINAL ARRAY OUTPUT is 960 WATTS EOL (3 YEARS). BATTERY LOSS IN ECLIPSE IS APPROXIMATELY 9.87 AMPERE-HOURS. THIS IS EQUIVALENT TO 14% DEPTH-OF-DISCHARGE FOR THE TWO 35 A-H BATTERIES, (OR 28% WITH ONE BATTERY FAILED).

* INCLUDES ONE 22-MINUTE DATA DOWNLINK.

GPS II POWER REQUIREMENTS SUMMARY, TOPEX

With the increased cell area recommended, (i.e., filling unused substrate area with cells), it appears that the GPS has more than adequate solar array power to meet the mission requirements for TOPEX. The added cells would weight 11.8 pounds; however, the NAV payload DC/DC converter would be eliminated, as it is not required for the TOPEX mission. It weighs 13 pounds, so there is a net weight change of -1.2 lb.; (see diagram of GPS II EPS in Task 5).

With a total of three batteries, there is no problem with battery reconditioning, in that there appears to be adequate power excess to charge two batteries at the same time, (i.e., early in the mission before too much array degradation, and when battery reconditioning would be required).

In that GPS II is designed for a mission life in excess of the 5 year TOPEX option, there does not appear to be a problem meeting the 5-year life with respect to power generation.

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GPS II POWER REQUIREMENTS SUMMARY FOR TOPEX

SUBSYSTEM/LOAD	DATA DOWNLINK (SUNLIGHT)	NO DOWNLINK (ECLIPSE)
C&DH/TDRS	154.9 WATTS (AVE)*	137.0 WATTS
ATTITUDE CONTROL	25.1	25.1
ELECTRICAL POWER	32.8	32.8
THERMAL CONTROL	10.0 (AVE)	10.0 (AVE)
SUB-TOTAL	222.8	204.9
10% POWER MARGIN	22.3	----
TOPEX PAYLOAD	259.0	259.0
TOTAL (P/L + SUBS. + MARG.)	504.1 WATTS	463.9 WATTS

BATTERY DRAIN DURING ECLIPSE = 0.58 HRS (34.7 MINS) X 463.9 WATTS = 269.1 WATT-HOURS

BATTERY CHARGING POWER NEEDED FOR MINIMUM TIME IN SUNLIGHT = 269.1 ÷ 1.29 HRS (77.3 MINS.) = 208.6 WATTS

ALLOWING FOR ENERGY TRANSFER AND CHARGIN INEFFICIENCY, 278 WATTS WOULD BE NEEDED TO RECHARGE THE BATTERIES IN 77.3 MINUTES. ADDED TO THE PAYLOAD AND SUBSYSTEM TOTAL, THE ARRAY DEMAND IS 782.1 WATTS. THE EXISTING GPS II ARRAY DESIGN HAS A NOMINAL OUTPUT OF 700 WATTS EOL (5 YEARS). HOWEVER, THERE IS UNUSED AREA ON THE SUBSTRATES WHICH, IF FILLED, WOULD INCREASE THE OUTPUT TO 900 WATTS EOL. BATTERY LOSS IN ECLIPSE IS APPROXIMATELY 10.6 AMPERE-HOURS. THIS IS EQUIVALENT TO 15% DEPTH-OF-DISCHARGE FROM TWO 35 A-H BATTERIES, WITH THE THIRD BATTERY BEING REDUNDANT.

* INCLUDES ONE 22-MINUTE DATA DOWNLINK

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P80-1 GN_2 AND HYDRAZINE REACTION CONTROL SUBSYSTEMS FOR TOPEX

Preliminary assessment of the two RCS systems used on the P80-1 indicates that no modification would be required for their use in the TOPEX mission. In both cases, the quantity of propellant appears adequate for the 3 year mission. Extending the mission out to 5 years may require additional GN_2 propellant, resulting in a modification for larger storage tanks.

An assessment has been made of the P80-1 0.2 lb_f thrusters as to their capability to perform the specified minimum orbital correction maneuver of 10 mm/sec., with accuracy of $\pm$ 1 mm/sec. No problem is expected with these thrusters accommodating this requirement if the thrusters are carefully matched (by the supplier) prior to their selection for installation.

GPS II REACTION CONTROL SUBSYSTEM FOR TOPEX MISSION

The GPS II RCS requires minor modification for the TOPEX mission application. The quantity of propellant necessary for orbit control exceeds the tank capacity of the GPS blowdown system. Initial assessment would add two GN_2 pressurant tanks to the system with propellant tanks loaded to near full capacity.

The GPS II RCS 0.1 lb_f thrusters will meet the requirement for 10 $\pm$ 1 mm/sec. easily, if the thrusters are carefully matched by the supplier prior to installation in the spacecraft's RCS modules.

P80-1 ATTITUDE CONTROL & DETERMINATION SUBSYSTEM

TOPEX MODIFICATIONS

The TOPEX pointing requirements are summarized on the facing table. The first 4 items are taken from Exhibit I of the Statement of Work, and the last item is derived from the TDRS antenna beamwidth of 12.3°. The omni-antenna for ground station communications imposes no ACDS requirements. Nadir pointing for the experiments and their 1σ values are relatively coarse and impose no stringent requirements. The requirement to point the TDRS antenna at the TDRSS satellite imposes a 3-axis attitude determination requirement on the ACDS. Again, however, this requirement is not exceptional compared to the state-of-the-art.

The P80-1 ACDS capability is more than adequate for the TOPEX mission requirements without modification. It is anticipated that further analysis would result in only minor modifications, if any, being necessary. Actually, the P80-1 ACDS is 'over designed' for TOPEX use, and some cost savings might be realized by eliminating some unnecessary components and simplifying the ACDS for TOPEX use. (This was not taken into consideration for the cost estimates.

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TOPEX ATTITUDE CONTROL AND DETERMINATION

REQUIREMENTS

UNIT	MODE	OPTION 1 (DEGREES)	OPTION 2 (DEGREES)	OPTION 3 (DEGREES)	ATTITUDE
RADIO ALTIMETER	CONTROL 1σ	0.15	0.25	0.25	NADIR
	DETERMINATION 1σ	0.05	0.10	0.10	NADIR
RADIOMETER		0.25	0.25	0.25	BORESIGHT
RADIOMETRIC TRACKING	CONTROL	5.0	5.0	5.0	NADIR
LASER RETROREFLECTOR	CONTROL	TBD	TBD	TBD	NADIR
TDRS ANTENNA	DETERMINATION	2.5	2.5	2.5	3-AXIS

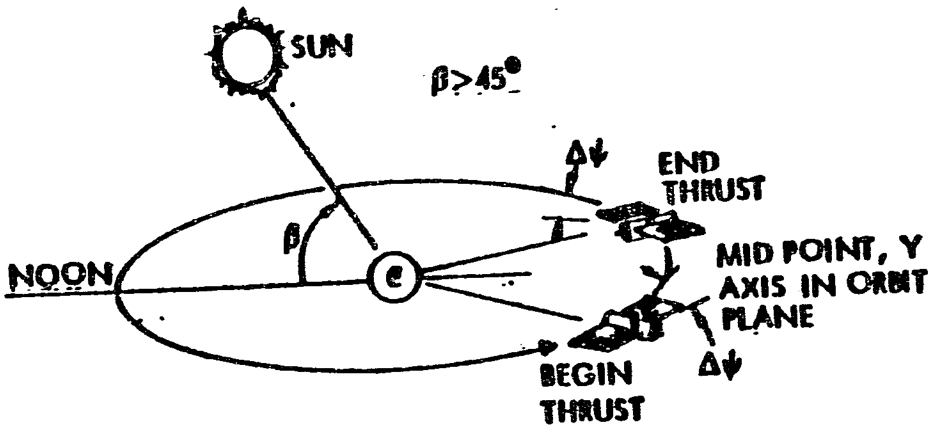
GPS II ATTITUDE & VELOCITY CONTROL/TOPEX MODIFICATIONS

The information on the facing chart is given to demonstrate that the AVCS on GPS II can perform delta-V maneuvers for any sun-line inclination to the orbit plane in contrast to GPS Phase I, in which a delta-V maneuver could only be performed near orbit noon or midnight for $\varphi > 45^\circ$. Consequently, delta-V maneuvers can be performed by GPS II on every orbit, if necessary. In addition, this figure illustrates the way in which the solar array sun sensors be used to estimate the yaw angle during no eclipse periods. The nominal yaw angle is a function of β and the position of the satellite in its orbit, and can be readily computed. The solar array sun sensors will measure the error signal and then can be summed to obtain the actual yaw angle. This procedure will give an accurate estimate of yaw near dawn and dusk for any φ. As $\varphi \rightarrow 0$, the estimate of yaw near noon will be useless if the raw data is used. In these situations, the yaw rate can be estimated by monitoring the reaction wheel momentum changes and comparing them to the nominal values for perfect control. During eclipse periods, the yaw rate can be controlled by driving the reaction wheels to interchange stored momentum as defined by the orbit characteristics and yaw rate profile required. The accuracy of this technique has not yet been determined. Clearly, software changes in the Control Electronics Assembly will be required to implement this yaw estimate technique, and possibly a more accurate sun sensor will be required.

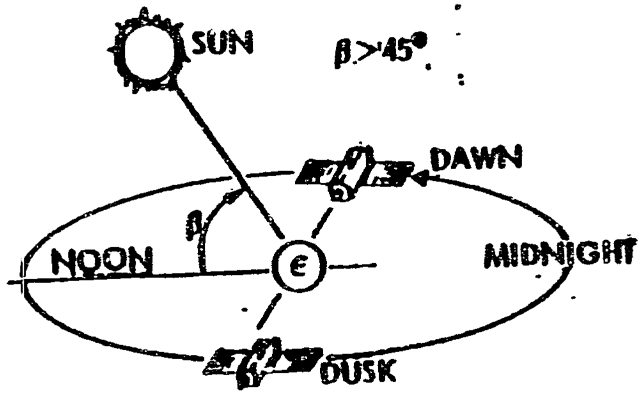
The GPS II AVCS Earth sensor will have to be changed out for a horizon sensor for low orbit application. Weight and power values would remain about the same as that for the basic mission.

GPS II ACDS/RCS SMALL THRUSTER DELTA-V CONCEPT
(STATION ACQUISITION, REPHASING AND STATIONKEEPING)

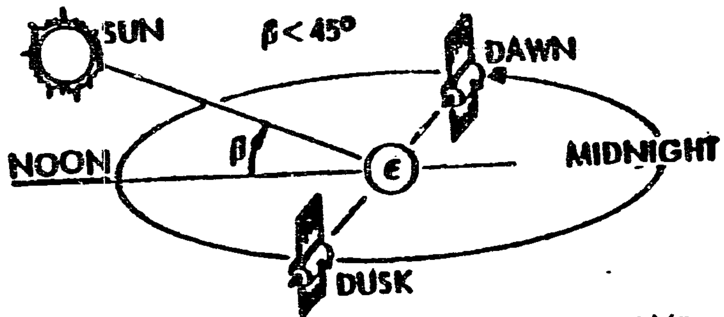
- ALLOWS DELTA-V DURING ECLIPSE SEASON
- MINIMIZES DELTA-V AND CONTROL THRUSTER IMPINGEMENT ON THE SOLAR ARRAYS



ARRAYS IN YZ PLANE (90°)
USE YAW (Y AXIS) THRUSTERS
YAW BIAS = 0



ARRAYS IN ORBIT PLANE USE
YAW (Y AXIS) THRUSTERS
YAW BIAS = 90-β



ARRAYS NORMAL TO ORBIT PLANE
USE X-TRANSLATION THRUSTERS
YAW BIAS = β

ATTITUDE CONTROL SUMMARY FOR TOPEX MISSION REQUIREMENTS

The P80-1 ACDS is more than adequate to meet the TOPEX requirements without modification.

The GPS II AVCS is not adequate as configured, but can be modified to meet the requirements. The primary changes are installation of a low Earth sensor (such as the Ithaco Universal Earth Sensor), and generation of a yaw reference for TDRS antenna steering. The preferred method would be to use the ephemeris data and solar array sun sensor outputs with estimation for low φ angles (near noon) in eclipse perids. Alternatively, the rate gyro assembly that is used only for telemetry could be incorporated into the control loop after skewing such that each of the three gyros measures a component of yaw as well as pitch and roll. This would be primarily for redundancy, since any one gyro would give the yaw value after using the Earth sensor output to remove the pitch and roll components. The gyros would still be capable of performing their telemetry functions as well. Anticipated growth and the low altitude yaw steering mode might dictate higher torque and momentum capability for the reaction wheels and higher dipoles for the electro-magnets. If so, this would require hardware modifications to the Control Electronics Assembly, as well as the software changes to implement the yaw steering mode.

ATTITUDE CONTROL SUMMARY FOR TOPEX MISSION

- P80-1 ATTITUDE CONTROL & DETERMINATION SUBSYSTEM - NO MODIFICATIONS NECESSARY
- GPS II ATTITUDE & VELOCITY CONTROL SUBSYSTEM:
  - REPLACE EARTH SENSOR WITH ITHACO UNIVERSAL EARTH SENSOR
  - GENERATE EXPLICIT YAW REFERENCE FOR TDRS ANTENNA STEERING
  - MODIFY CONTROL ELECTRONICS ASSEMBLY SOFTWARE (FOR YAW REF.) AND POSSIBLY SOME DRIVE ELECTRONICS HARDWARE.
  - POSSIBLY INCREASE SIZE OF REACTION WHEELS AND/OR ELECTRO-MAGNETS (P80-1 ELECTRO-MAGNETS WOULD BE QUITE SATISFACTORY) FOR MORE TORQUE AND MOMENTUM CAPABILITY

P80-1 THERMAL CONTROL/TOPEX MODIFICATIONS

No major modifications for the Thermal Control Subsystem of the P80-1 spacecraft are envisioned. It is expected that some equipment and their radiators will have to be relocated to accommodate the TOPEX payload. Without the Teal Ruby telescope, the heat rejection capability is increased.

With the TOPEX payload information given in Exhibit 1 of the Statement of Work, there appears to be no problem accommodating the thermal control of the TOPEX payload components, either outside or inside the spaceframe. From the data available from the JPL PHASE A FINAL REPORT, all internally mounted units can be kept within their operating and non-operating ranges by passive thermal control methds.

It can be reasonably assumed that the weight for the P80-1 TOPEX Thermal Control Subsystem will be about the same as that for the P80-1 basic mission, 55 lb. Heater power would remain at about 100 w.

As explained in Task 5, P80-1 uses no special thermal protection in the Shuttle cargo bay after bay doors are open, in that it is the first payload ejected. If this were not the case for P80-1 TOPEX, some thermal shielding might be required to prevent overheating (in direct sunlight) or getting too cold (facing deep space). This could be accommodated.

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GPS II THERMAL CONTROL/TOPEX MODIFICATIONS

No extensive modifications would be required of the GPS II Thermal Control Subsystem to accommodate the TOPEX mission. Some redesign of the thermal doublers on the equipment shelf to accommodate the TOPEX payload and the C&DH/TDRS components would be expected. No difficulty is anticipated in providing thermal control protection to the new/added components for the TOPEX mission. At this time, it appears that passive techniques can be used for providing the required temperature limits inside the spaceframe, and for maintaining reasonable temperatures (and temperature gradients) to components mounted outside the spaceframe.

Some modification of the ASE sun-shield might be required to allow for protrusions of the TOPEX payload on the GPS II forward bulkhead, (see diagram under Thermal Control in Task 5).

The Thermal Control subsystem weight and power would be about the same as that for the basic GPS II mission, 142 lb. and 27 watts.

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Rockwell International

Space Operations/Integration &
Satellite Systems Division

Task 5

"Provide descriptions of subsystems including subsystem performance characteristics."

Descriptions and performance characteristics for the subsystems of the P80-1 and GPS Phase II candidates are presented in the test and pictorials of this task response.

- Structure
- Telemetry, Tracking & Command
- Electrical Power
- Reaction Control
- Orbit Insertion Propulsion
- Attitude Determination & Control
- Thermal Control

Descriptions and performance characteristics of the P80-1 and GPS Phase II Shuttle Orbiter cargo bay and aft flight deck Aerospace Support Equipments (ASE) has been included in this task response.

PSO-1 STRUCTURE SUBSYSTEM DEFINITION

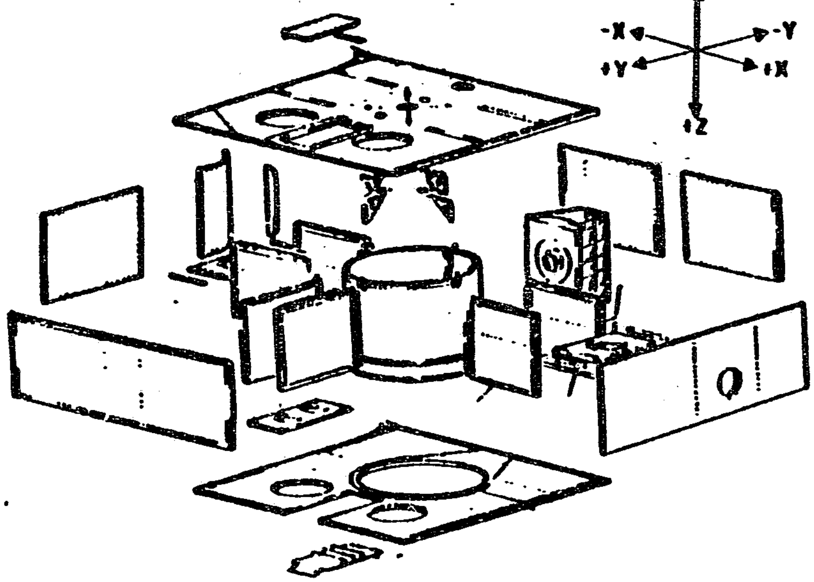
The PSO-1 Structure Subsystem consists of the basic spaceframe, the solar array substrates, and all the required bracketry and fasteners for the subsystems and experiments. The basic spaceframe is a simple box-frame utilizing aluminum honeycomb paneling, yet it has been designed (with more than adequate margin) to support an 825 pound Teal Ruby electro-optical telescope and its steerable yoke on one side of the spaceframe, and a 384 pound single-degree-of-freedom solar array on the opposite.

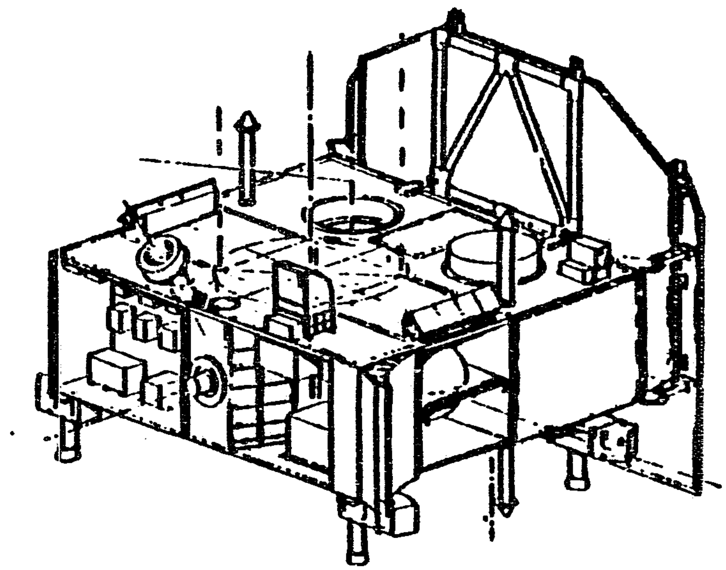
The structure has also been designed to withstand the Shuttle Orbiter dynamic launch acoustic environment qualification level of 150.dB Overall Sound Pressure Level (OASPL), and 3 axis accelerations and OIS motor burn.

The basic structure design approach and the assembled structure configuration are shown in the facing pictorial diagram.

P80-1 STRUCTURE SUBSYSTEM DESCRIPTION

- ALUMINUM HONEYCOMB PANELS
- MONOCOQUE THRUST CYLINDER





- SUBSYSTEM INSTALLATION
  - OPTIMIZED TO
    - DYNAMICS/LOADS
    - WEIGHT DISTRIBUTION
    - THERMAL
    - HUMAN FACTORS

STRUCTURE WEIGHT: 729.2 LB. (331.5 KG) (INCL. SOLAR PANEL SUBSTRATES AND ALL COMPONENT/EXPERIMENT ATTACHMENT BRACKETRY AND FASTENERS).

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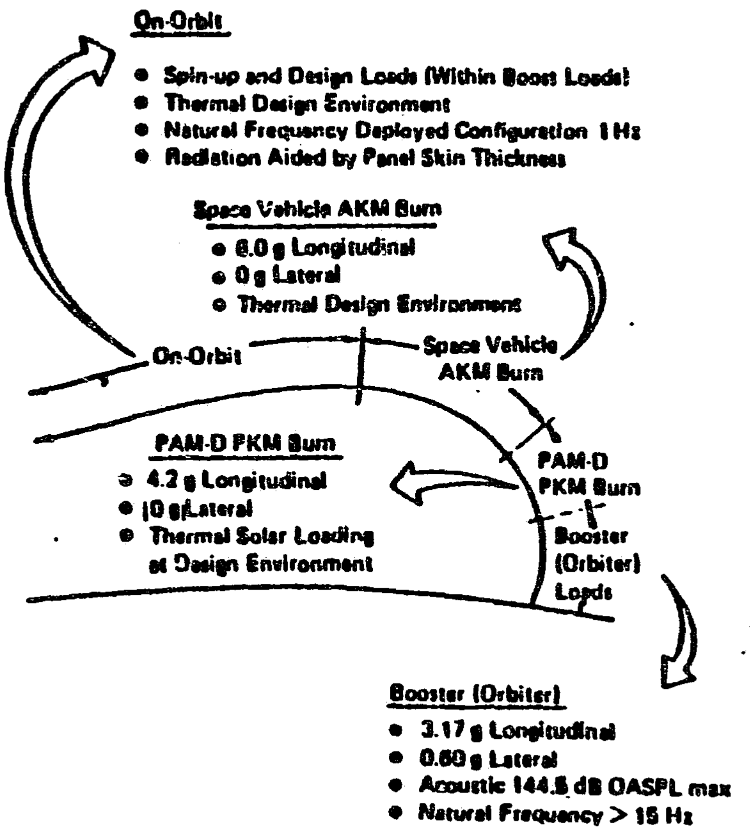
GPS II STRUCTURE SUBSYSTEM DEFINITION

The selected structural concept for GPS II provides a lightweight spaceframe that withstands launch and orbital loads and supports/protects the subsystem and payload components on its bulkheads and fixed shear panels. The design rationale used for the structure includes minimum cost, minimum weight, reliability, legacy (heritage) from the GPS I design, and maximum use of off-the-shelf hardware. The design not only meets current mission objectives, but also has growth potential.



Some of the structural features are:

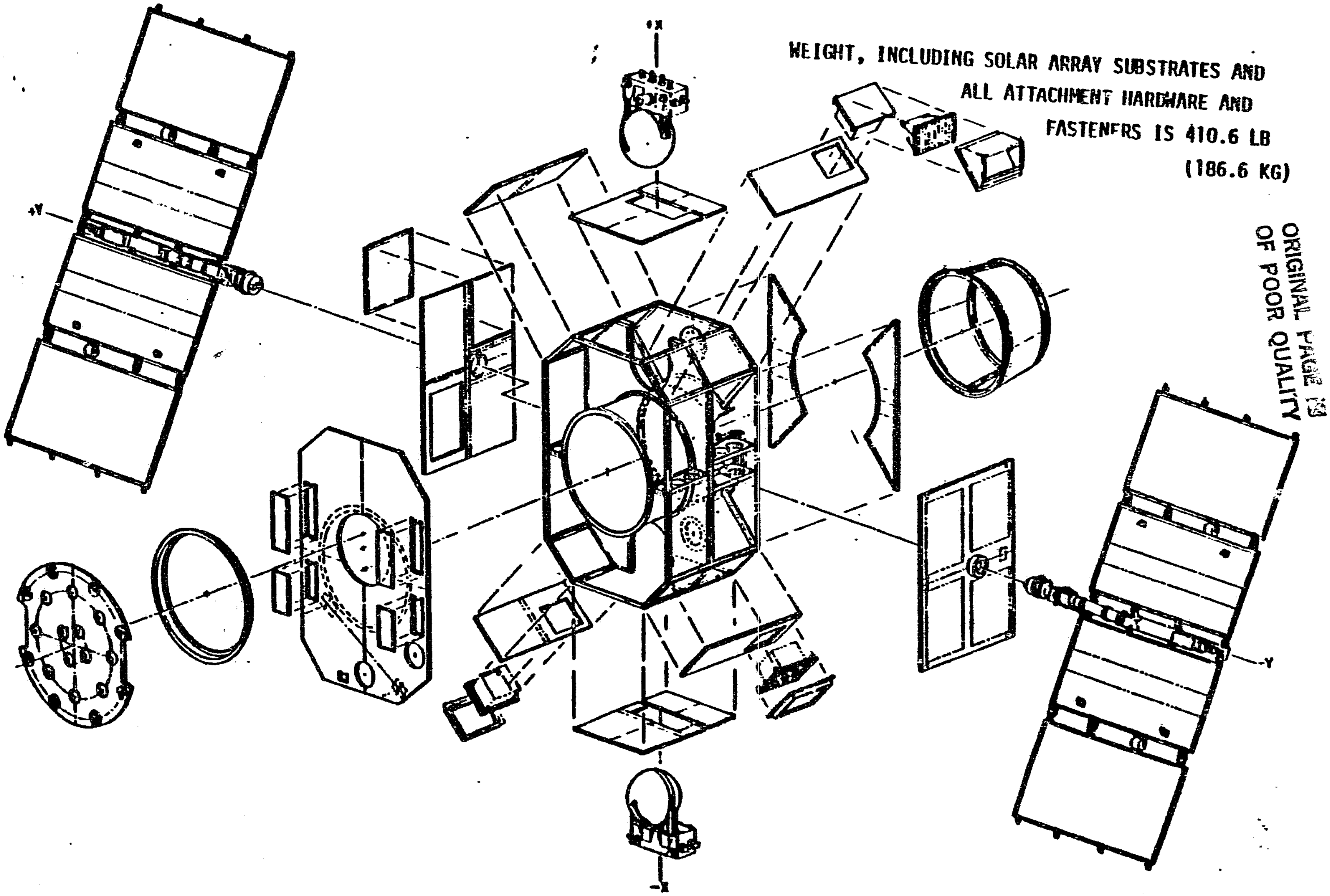
1. Lightweight rigid spaceframe of aluminum-bonded honeycomb and has a natural frequency of greater than 15 Hertz.

2. X-axis side panels are removable for internal access.

3. Driven by the acoustic environment of a Shuttle launch, preliminary analyses were performed to establish adequate margins and provide qualification/acceptance levels to subcontractors.

The diagram at the left identifies the various structural requirements at various phases of the mission; and the pictorial diagrams on the facing page describes the basic GPS II structure. Weight statements for both PBO-1 and GPS II are given on subsequent pages.

GPS II STRUCTURE SUBSYSTEM DESCRIPTION



WEIGHT, INCLUDING SOLAR ARRAY SUBSTRATES AND ALL ATTACHMENT HARDWARE AND FASTENERS IS 410.6 LB (186.6 KG)

P80 AND GPS II WEIGHT STATEMENTS

The weights of both spacecraft through various phases of the mission are shown in the facing table. It should be noted that GPS II has added considerable weight throughout its subsystems for protection against radiation. In comparing lift-off weight to payload weight, it is noted that GPS II is orbited as half-synch, while the orbital altitude of P80-1 is 400 n.mi.

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P80-1 AND GPS II WEIGHT STATEMENTS

WEIGHT ADDED OR REMOVED	P80-1	GPS II	
PAYLOAD	1,568.3	312.3	
STRUCTURE	792.2	410.6	
BALLAST	104.9	11.8	
ELECTRICAL POWER	368.6	290.6	
POWER/SIGNAL WIRE HARNESSES	233.0	191.8	(GPS II WT. INCL. 68.4 LB. FOR RADIATION HARDNESS)
REACTION CONTROL (DRY)	230.0	51.8	
TELEMETRY, TRACKING AND COMMAND	199.6	79.4	
ATTITUDE CONTROL	228.3	87.9	
THERMAL CONTROL	55.0	142.0	
SPACECRAFT (DRY)	3,549.9	1,578.2	
RCS PRESSURANT/PROPELLANT	75.0	93.0	
APOGEE INSERTION SRM EMPTY CASE	---	148.0	(P80-1 JETTISONS ALL MOTOR CASES)
INITIAL ON-ORBIT	3,624.9	1,819.2	
APOGEE INSERTION SRM/PROPELLANT	2,475.7	1,929.5	
POST-PERIGEE INSERTION	6,100.6	3,748.7	
PERIGEE INSERTION SRM	2,475.7	4,658.0	
SHUTTLE SEPARATION	8,576.3	8,406.7	
SHUTTLE AEROSPACE SUPPORT EQUIPMENT	2,117.0	2,555.0	
CHARGEABLE LIFT-OFF WEIGHT	10,693.3 LB	10,961.7 LB	

P80-1 SPACECRAFT TELEMETRY, TRACKING AND COMMAND SUBSYSTEM

TELECOMMUNICATIONS AND PHYSICAL CHARACTERISTICS

The facing pictorial shows the Telemetry, Tracking and Command (TT&C) subsystem components, their general interrelationship, and a brief description of the basic telemetry and command design capabilities. The P80-1 has more hardware than the GPS II TT&C subsystem because of a larger payload data and data storage (tape recorders) requirement.

The components shown include command decryptors and data encryptors which would not be required for unclassified missions. Most of the P80-1 components are hard-wired unique designs which will require modification for other applications.

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P80-1 TELEMETRY, TRACKING & COMMAND SUBSYSTEM CHARACTERISTICS

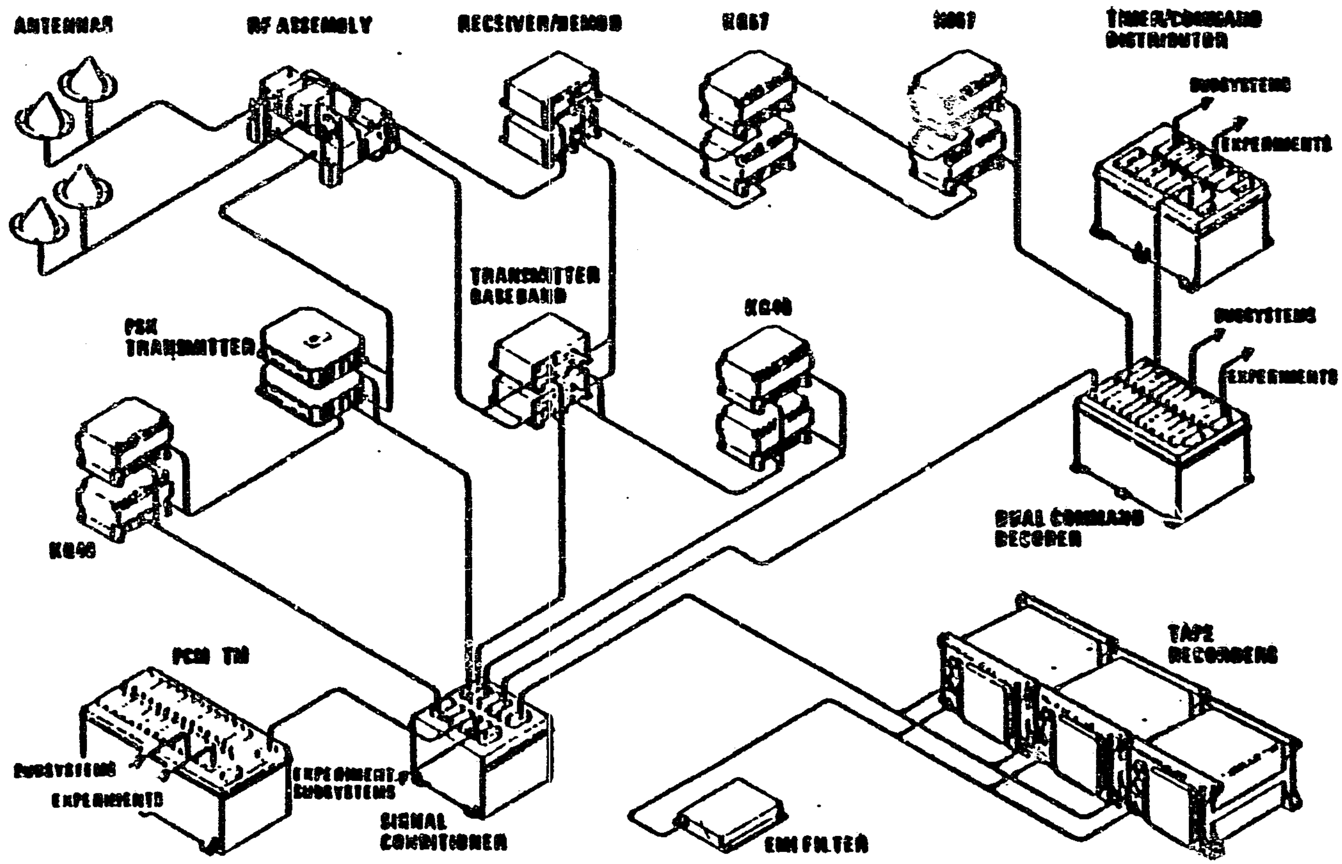
- DESIGNED FOR COMPATIBILITY WITH AFSCF
- TWO TELEMETRY LINKS
  - CARRIER 1

    32 KBPS NRZ-L
    2212.5 MHz (1.024 MHz SUBCARRIER)
    ENCRYPTED
  - CARRIER 2

    1.024 MBPS NRZ-L
    2207.5 MHz
    ENCRYPTED
- TRACKING
  - SGLS COMPATIBLE
- COMMAND
  - 42 BIT WORD 1771.729 MHz
  - 2K BAUD
  - 1, 0 & S TONES
  - ENCRYPTED

WEIGHT: 199.6 LB (90.7 KG)

POWER: 232.5 WATT MAX.



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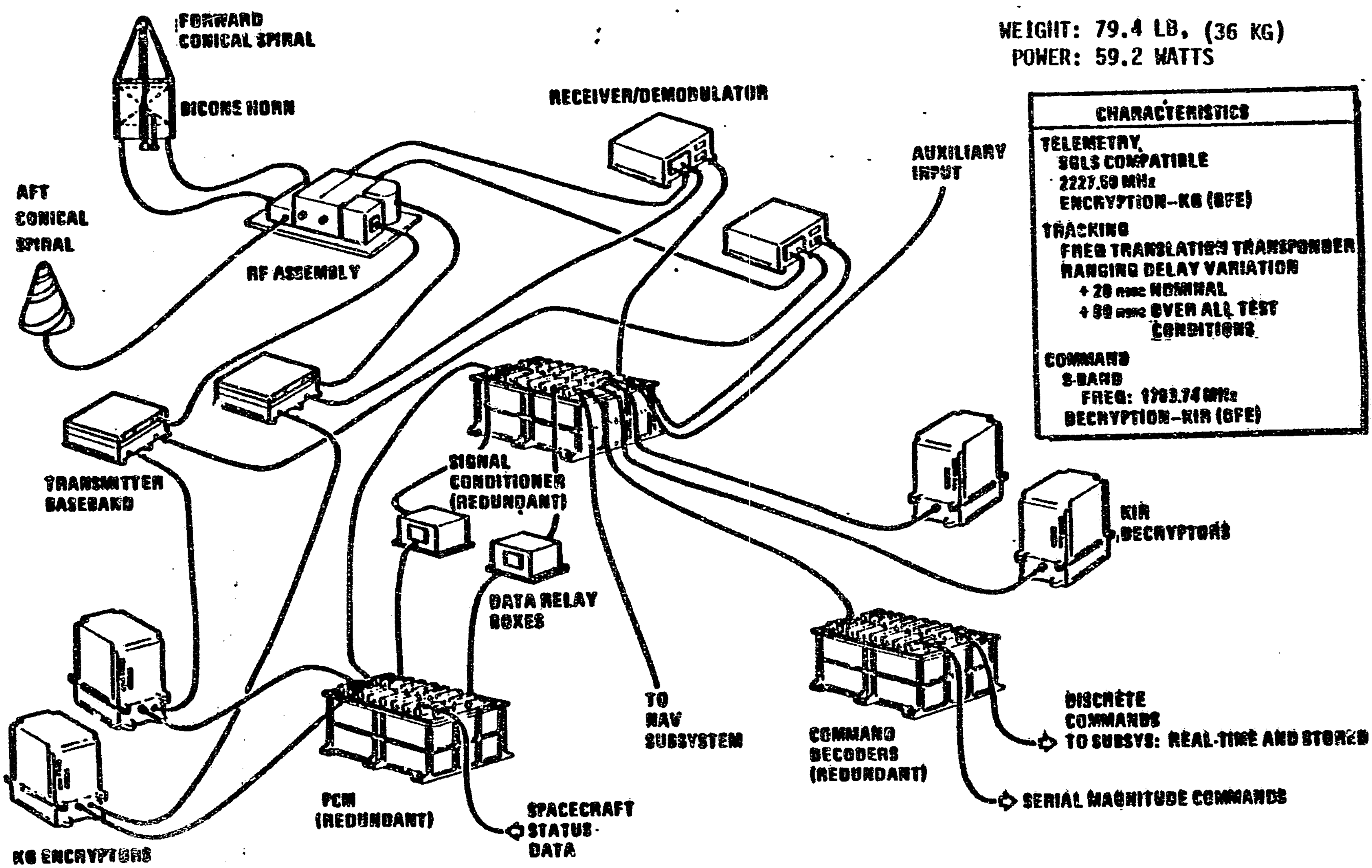
GPS II SPACECRAFT TELEMETRY, TRACKING AND COMMAND SUBSYSTEM

TELECOMMUNICATIONS AND PHYSICAL CHARACTERISTICS

The facing pictorial illustrates the components and their interrelationships and a brief description of the telecommunications capabilities. The term AUXILIARY INPUT refers to data from the GPS II classified secondary payload. The primary payload, the Navigation Subsystem, is an RF navigation signal generator; thus, no primary payload data (other than housekeeping) is transmitted through the TT&C.

Data encryptors and command decryptors are used which would not be required on an unclassified mission. The components used for the GPS II TT&C are hard-wired unique designs which would require modification for other applications.

GPS II TELEMETRY, TRACKING & COMMAND SUBSYSTEM CHARACTERISTICS



P80-1 AND GPS II TT&C TELEMETRY AND COMMAND CAPABILITIES

The data and command capacities for both the P80-1 and the GPS II TT&C subsystems are shown on the facing table. Both serial and discrete command capability is available in both subsystems. The data format has been specifically designed for the Air Force Satellite Control Facility (AFSCF) use. The use of hard-wired devices inherently leads to a rather inflexible data format and therefore would require extensive modification for other applications.

P80-1 & GPS TT&C CAPABILITIES

SYSTEM	MEASUREMENT			COMMAND	
	ANALOG	BILEVEL	SERIAL	DISCRETE	SERIAL
ACDS	25 ① /57 ②	33/52	3/2	18/103	1/1
EPS	30/33	35/134	—	76/234	—/0
RCS/OIS	34/ —	8/10	—	26/ —	—/ —
TCS	78/32	20/10	—	35/ —	—/ —
C&DH	15/42	57/126	3/9	57/115	2/2
GPS P/L	65/ —	71/ —	3/ —	124/ —	4/ —
INSTRUMENTS	—/51	—/47	—/6	—/126	—/5
TOTALS	314/215 (47 SPARES)	224/379 (88 SPARES)	9/17 (2 SPARES)	336/578 (48 SPARES)	7/8 (2 SPARES

① GPS II CAPABILITIES

② P-80 CAPABILITIES

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P80-1 ELECTRICAL POWER SUBSYSTEM PHYSICAL DESCRIPTION AND POWER CAPABILITIES

The P80-1 Electrical Power Subsystem (EPS) is of a fairly typical configuration except for its single solar array. This was established by the Teal Ruby Experiment, a large electro-optical sensor telescope which is mounted on a pivotal yoke on the side of the spacecraft opposite the solar array. The array nominally produces 950 watts EOL, and requires a 180 degree yaw maneuver every six months.

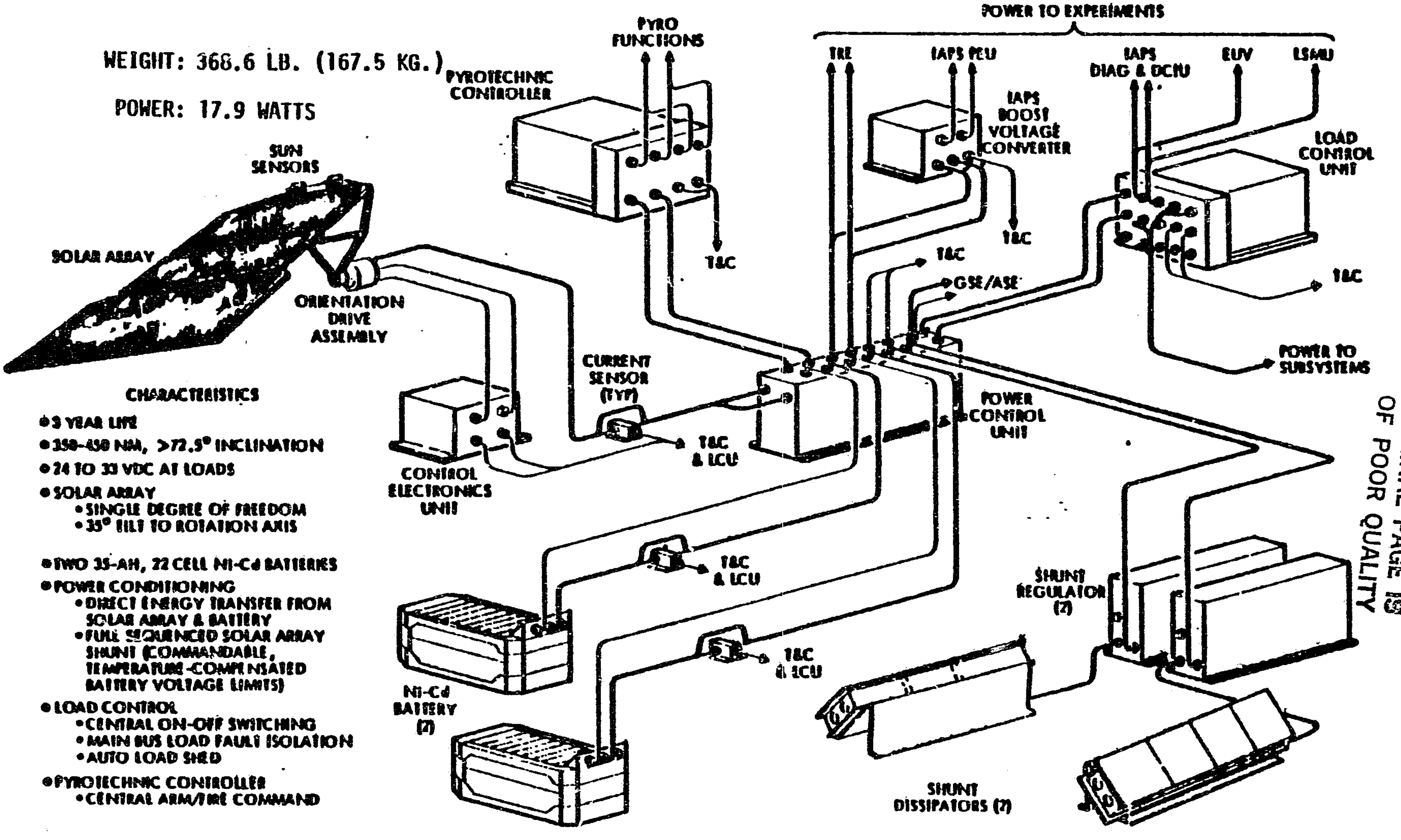
The facing pictorial shows the EPS components and their interrelationships. The weight of the subsystem includes all solar cells but does not include the weight of the solar panel substrates (which is charged to the Structure Subsystem). The EPS weight does not include the weight wire harnesses.

The low power requirement of the EPS is the result of its Direct Energy Transfer design.

P80-1 ELECTRICAL POWER SUBSYSTEM CAPABILITIES AND PHYSICAL CHARACTERISTICS

WEIGHT: 368.6 LB. (167.5 KG.)

POWER: 17.9 WATTS



CHARACTERISTICS

- 3 YEAR LIFE
- 350–450 NM, >72.5° INCLINATION
- 24 TO 33 VDC AT LOADS
- SOLAR ARRAY
  - SINGLE DEGREE OF FREEDOM
  - 35° TILT TO ROTATION AXIS
- TWO 35-AH, 22 CELL Ni-Cd BATTERIES
- POWER CONDITIONING
  - DIRECT ENERGY TRANSFER FROM SOLAR ARRAY & BATTERY
  - FULL SEQUENCED SOLAR ARRAY SHUNT (COMMANDABLE, TEMPERATURE-COMPENSATED BATTERY VOLTAGE LIMITS)
- LOAD CONTROL
  - CENTRAL ON-OFF SWITCHING
  - MAIN BUS LOAD FAULT ISOLATION
  - AUTO LOAD SHED
- PYROTECHNIC CONTROLLER
  - CENTRAL ARM/FIRE COMMAND

GPS II ELECTRICAL POWER SUBSYSTEM PHYSICAL DESCRIPTION AND POWER CAPABILITIES

The GPS II Electrical Power Subsystem (EPS) is of a typical configuration and therefore has fairly general application. The components of the EPS and their interrelationships are shown on the facing pictorial.

The weight of the subsystem includes all the solar cells but does not include the solar array substrates. These, and the subsystem's wire harness weights, are charged elsewhere.

It should be noted that the stated wattage output of the solar arrays is based on the solar cells required to conduct the GPS II mission and do not cover the entire solar panel area. If the solar panel substrates were completely covered with cells, the total output would exceed 900 watts EOL.

GPS II ELECTRICAL POWER SUBSYSTEM CAPABILITIES AND PHYSICAL CHARACTERISTICS

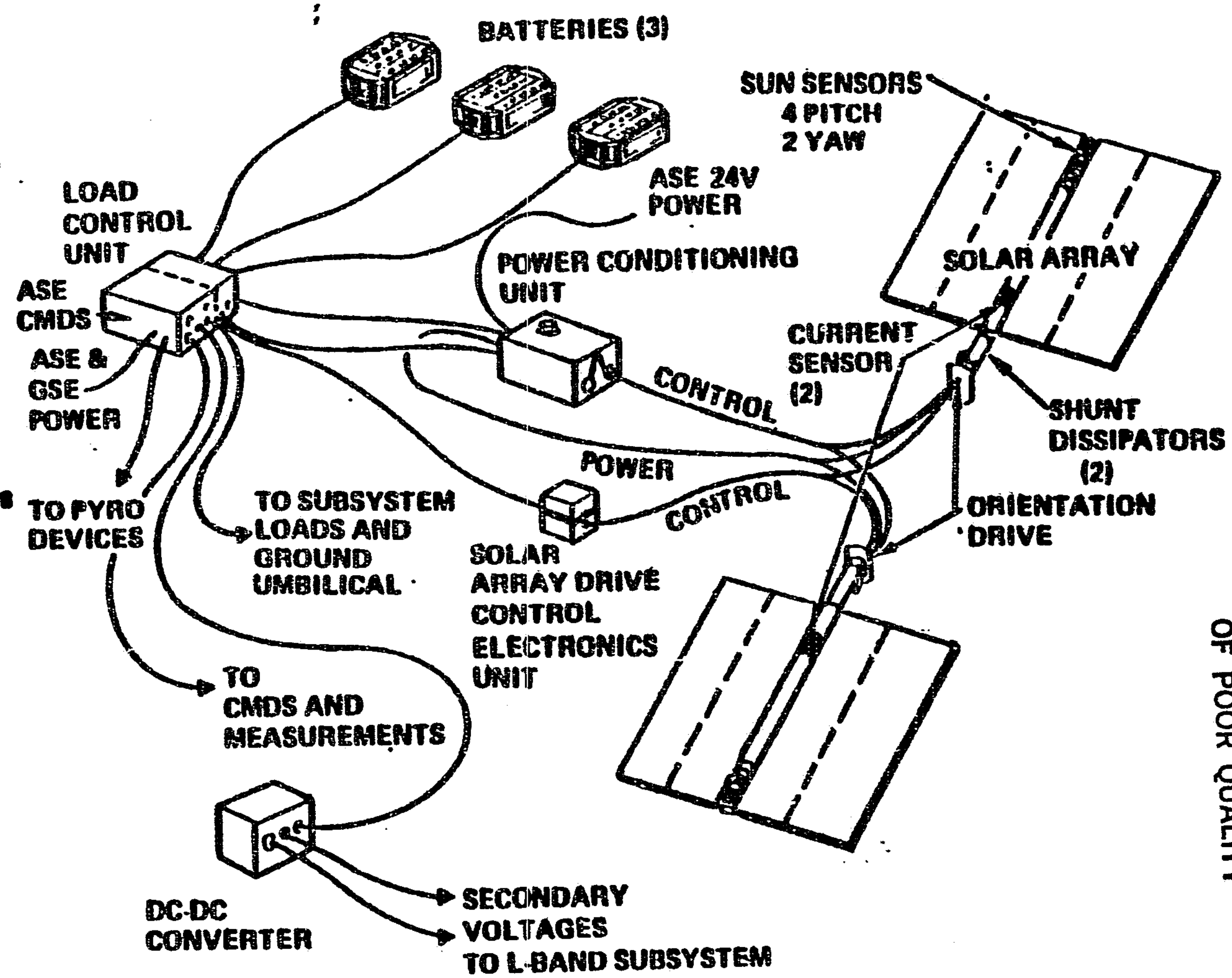
PERFORMANCE
- 27.8 ± 1.8 VOLT MAIN BUS
- 0.5 VOLT MAX DIST LOSS
- 700 W EOL SOLAR ARRAY (78 FT^2 AREA)
- 105 AMP HOUR STORAGE CAPACITY (3 NiC_d BATTERIES)
- AVERAGE SV LOAD – 614 WATTS
- GENERATE SECONDARY VOLTAGES FOR LBS SUBSYSTEM

POWER DISTRIBUTION
- ON/OFF LOAD CONTROL
- AUTO LOAD SHED OF NON CRITICAL LOADS
- BATTERY & GROUND POWER CONTROL
- MAIN BUS LOAD FAULT PROTECTION
- PYROTECHNIC DEVICE CONTROL
- SELECTED LOAD CURRENT MONITORING MONITORING

POWER REGULATION & CONTROL
- DEDICATED BATTERY CHARGERS (3)
- BATTERY BOOST CONVERTERS 3 REDUNDANCY
- AUTOMATIC CHARGE DISCHARGE CONTROL
- LOAD SHED DETECTION

WEIGHT: 290.6 LB (132 KG)

POWER: 32.8 WATTS



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P80-1 REACTION CONTROL SUBSYSTEM (RCS) PHYSICAL CHARACTERISTICS AND CAPABILITIES

There are two (2) Reaction Control Subsystems (RCS) in the P80-1 spacecraft: a cold-gas (GN_2) RCS and a monopropellant hydrazine RCS. The thrusters of the two subsystems are mounted in four modules at the perimeter of the bottom of the spaceframe. The GN_2 RCS provides the impulse requirements to maintain three-axis attitude control immediately after Shuttle Orbiter separation. GN_2 thrusters are required because the P80-1 is not spin-stabilized and operation of hydrazine thrusters in the close vicinity of the Shuttle Orbiter is not allowed. The GN_2 RCS also provides attitude control to the spacecraft through to turn-on of the Attitude Determination and Control Subsystem's inertial wheels. The GN_2 RCS also acts as a back-up to the ACDS over the duration of the entire mission.

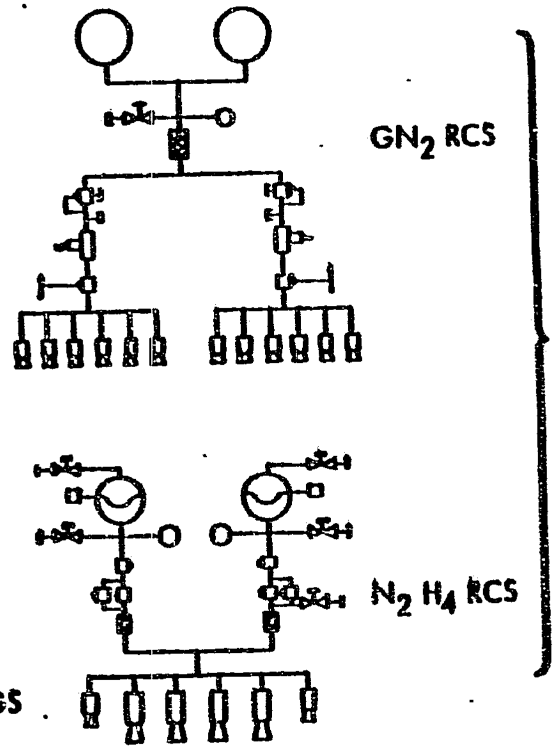
During solid motor firing, thrust vector control is provided by the higher thrust hydrazine RCS. Upon completion of the Apogee Insertion stage and its subsequent motor case jettison, the hydrazine RCS is isolated and the wetted lines are depleted or propellant.

The facing pictorial shows the simplified schematics of both Reaction Control Subsystems. Both subsystems are described in greater detail on subsequent charts.

- PROVIDE VEHICLE ATTITUDE CONTROL DURING

SHUTTLE SEP THROUGH WHEEL TURN-ON ON ORBIT (BACK UP)

- PROVIDE ATTITUDE CONTROL DURING SOLID MOTOR FIRINGS



P80-1 GN2 REACTION CONTROL SYSTEM

The subsystem consists of two (2) 12.55 inch diameter storage tanks, manifolded together, and connected through a 10-micron filter to two (2) redundant banks of regulators, isolation valves, relief valves and thrusters. This arrangement of components allows for flexibility in isolation of a redundant thruster bank should the need arise. The location of the isolation valves downstream of the regulators permits system reactivation without pressure overshoot.

The tanks are pressurized to 3700 psia, which is then regulated down to the working pressure of 60 psia. Each tank contains 10 pounds of GN_2. Thus, the "wet" weight of the subsystem is 72 lbs.

P80-1 GN_2 REACTION CONTROL SUBSYSTEM

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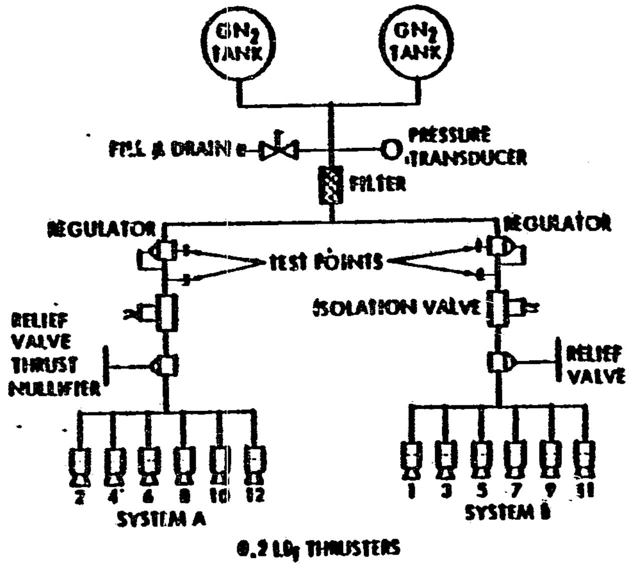
- GASEOUS NITROGEN (TWELVE .2 LBF THRUSTERS)
- PROVIDES ATTITUDE CONTROL
  - TRANSFER ORBIT THROUGH WHEEL TURN-ON
  - ON ORBIT AS BACKUP TO WHEELS

CHARACTERISTICS

COLD GAS (GN_2) REGULATED SYSTEM

TOTAL IMPUSE	≃ 1300 LB SEC
THRUST	0.2 LBS
I_{sp} SS	65 SEC
OPERATING PRESSURE	≃ 3700/60 PSIA
INERT WEIGHT	≅ 52 LBS
REDUNDANT THRUSTERS	
DUAL SEAT REGULATORS/VALVES	



P80-1 HYDRAZINE REACTION CONTROL SUBSYSTEM DESCRIPTION

The hydrazine RCS consists of four (4) 100 lb_f monopropellant thrusters and two (2) 5 lb_f monopropellant thrusters, and are mounted in four peripheral modules on the bottom of the spaceframe to provide three-axis steering of the spacecraft during solid rocket motor firings. The N_2H_4 is pressure fed to the thrusters from two 16.5 inch diameter positive expulsion tanks, each containing 27.5 pounds of hydrazine, in a blowdown mode, using GN_2 as the pressurant. Explosive pyrotechnic valves are used to control initial and final propellant flow start/stop of the thrusters. A filter is placed upstream from the thrusters. The thrusters are identical in design to those used in the Voyager spacecraft program.

At an operating pressure range of 450 to 250 psia, the corresponding thrust levels are 144 to 90 lb_f for the "100 lb_f" thrusters and 7 to 5lb_f for the "5 lf_b" thrusters.

The "wet" weight of the hydrazine RCS is approximately 122 lb_m.

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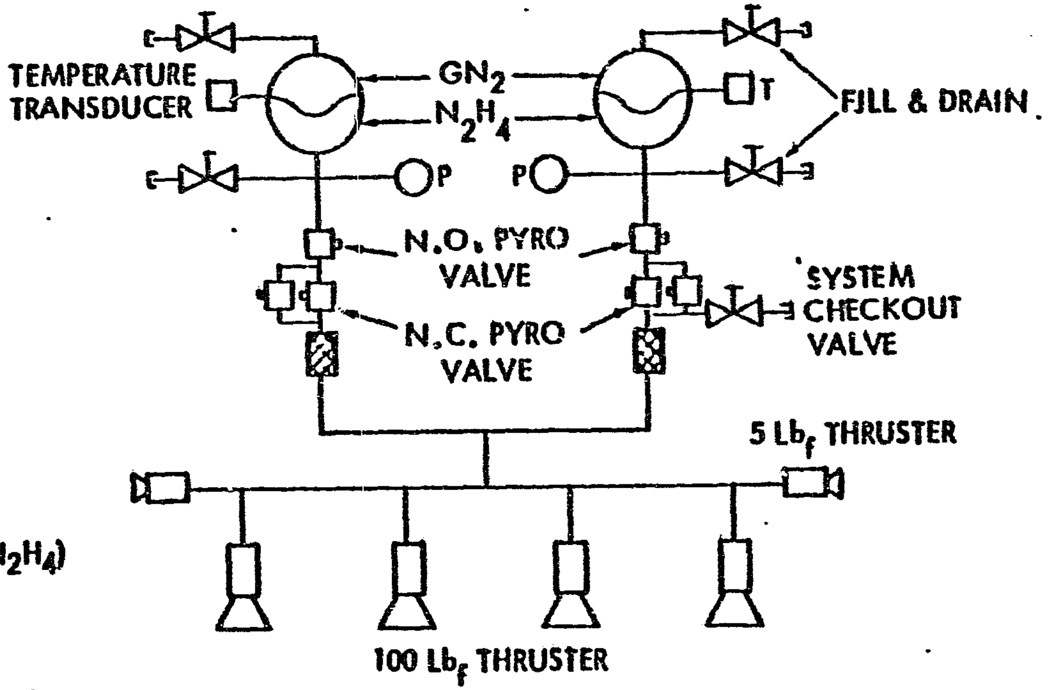
P80-1 N_2H_4 REACTION CONTROL SUBSYSTEM



CHARACTERISTICS

HYDRAZINE (N_2H_4) BLOW DOWN SYSTEM
TOTAL IMPULSE $\simeq$ 11,000 LB-SEC (55 LB_m N_2H_4)

THRUST P&R	144 → 90 LB_f
Y	7 → 5 LB_f
I_{SP} SS	220 SEC-MIN
OPERATING PRESSURE	450 → 250 PSIA
INERT WEIGHT	$\cong$ 65 LB

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GPS II REACTION CONTROL SUBSYSTEM DEFINITION

The GPS Phase II Reaction Control Subsystem (RCS) is a pressure blowdown monopropellant hydrazine system. The subsystem consists of a Propellant/ Pressurant Storage (PPS) assembly, a Propellant Distribution and Control (PDC) assembly, and the Rocket Engine Assembly Module (REAM). The PPS components consists of two (2) 6AL-4V titanium tanks measuring 16.5 inches in diameter, and each capable of being loaded with 47.5 pounds of hydrazine, and each containing an AF-E-332 hemispherically shaped diaphragm; two (2) propellant fill/drain valves; two (2) pressurant fill/drain valves; and two (2) temperature transducers. The PDC components consist of two (2) pressure transducers, two (2) filters, and four (4) latching isolation valves. The REAM components consist of twenty (2) 0.1 lb_f thrusters and two (2) 5 lb_f thrusters, each equipped with a temperature sensor.

Each bank of thrusters is mounted in its own module and the two modules are unted at the midpoint on opposite sides of the spacecraft. The 5 lb_f thrusters are mounted with opposing thrust vectors, which are parallel with the spin axis of the spacecraft. They are used in the pulse mode for attitude control maneuvering for proper placement of the spacecraft prior to firing the apogee motor, and in the steady-state mode for orbit trim/circularization. Both of these modes are executed while the spacecraft is in its spin mode.

The facing pictorial shows the GPS II RCS and its component relationships.

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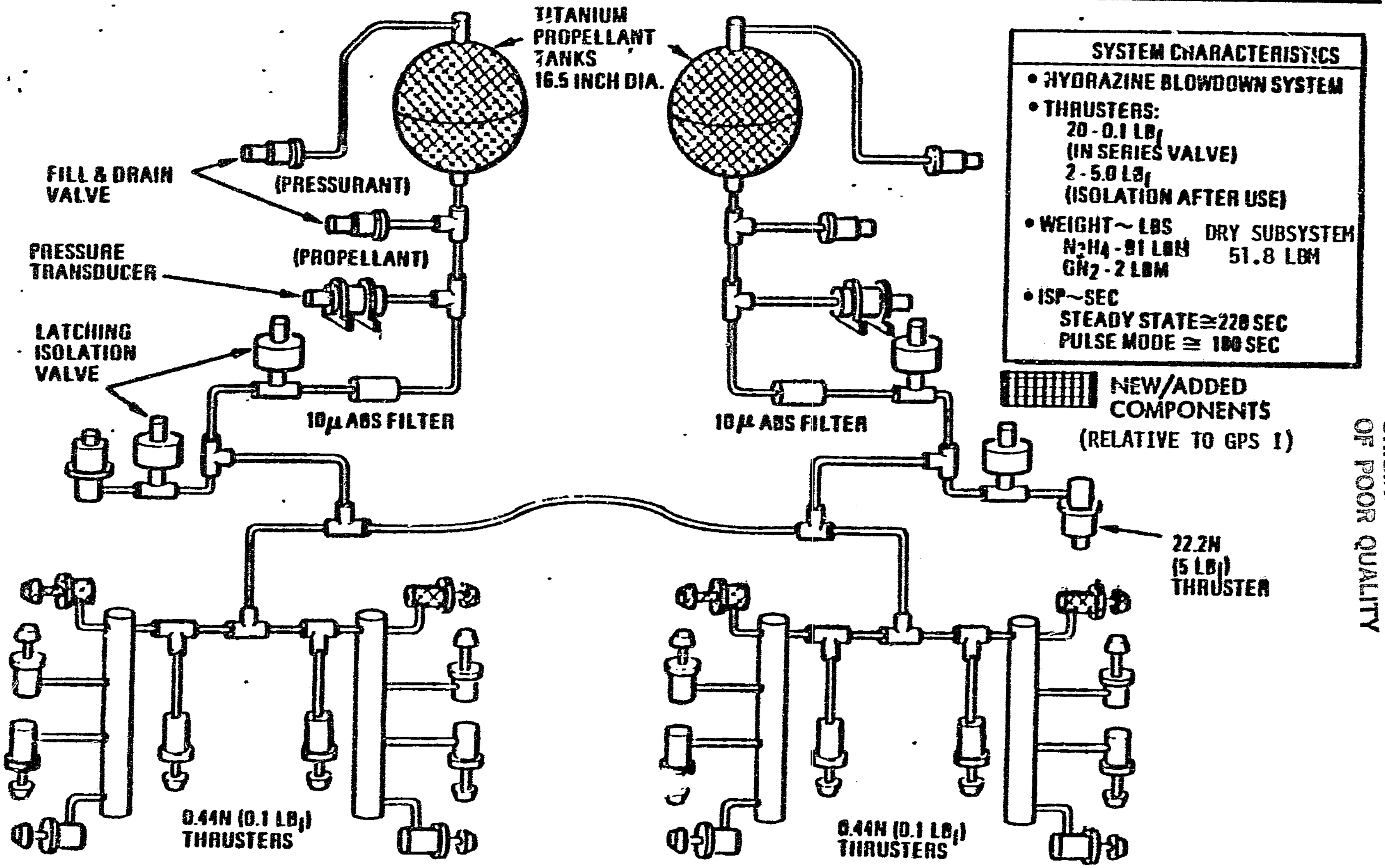
GPS II REACTION CONTROL SUBSYSTEM DESCRIPTION



P80-1 ORBIT INSERTION (SRM) SUBSYSTEM (OIS)

The ascent propulsion for the P80-1 spacecraft consists of two (2) tandem TEM-364-4 solid rocket motors. The first stage transfers the satellite from the Shuttle Orbiter parking orbit (160 n.mi.) to 400 n.mi. with some plane change. After burn-out, its empty case is jettisoned, and the second stage completes the necessary plane change and circularizes the orbit, after which its empty case is also jettisoned. The motors are identical and have been fully qualified.

Motor ignition is accomplished by an electromechanical safe and arm device, redundant explosive transfer lines and a Pyrogen igniter. Power to fire the motors is provided by the spacecraft batteries, through redundant circuitry, to each motor.

The facing pictorial details the performance characteristics of the motors.

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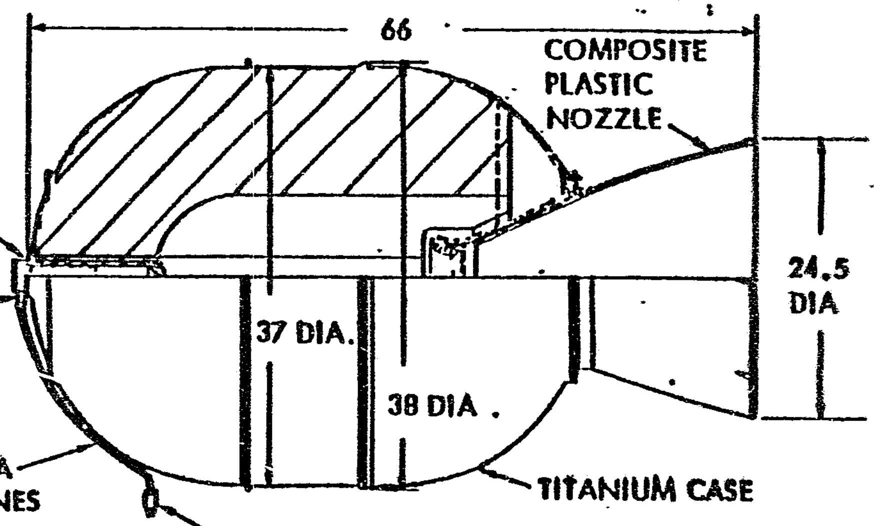
P80-1 SOLID ROCKET MOTOR DESCRIPTION

TE-M-364-4 MOTOR



MOTOR PERFORMANCE

TOTAL IMPUSE	654,360 LB-SEC
MAX THRUST	16,590 LB
ACTION TIME	42.92 SEC
PROPELLANT SPECIFIC IMPULSE	285.67 SEC
WEIGHT	
TOTAL LOADED	2475.7 LBS*
PROPELLANT	2290 LB
BURN OUT	169 LB
PROPELLANT	
DESIGNATION	TP-H-3062
FORMULATIONS	AP 70%
	AL 16%
	HYDRO CARBON BINDER 14%

*INCLUDES ESTIMATED WEIGHTS FOR S&A AND FLEXIBLE ETA'S

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GPS II PERIGEE STAGE DEFINITION

The perigee insertion stage is provided by the McDonald-Douglas Payload Assist Module (PAM-D) which is furnished to Rockwell as GFE by the Air Force. The PAM-D contains a STAR 48 solid rocket motor, and is shown in the facing pictorial.

The STAR 48 was subjected to a development/qualification program which utilized eight (8) motors loaded with from 3840 to 4400 pounds of propellant and nozzle expansion ratios of from 35 to 40. All eight were fired at the Arnold Engineering Development center under simulated vacuum conditions.

The STAR 48 has a thin-walled titanium case and a nozzle which contains a carbon/carbon exit cone. Key features of the motor assembly are:

- Full head-end web design
- 89% solid HTPB propellant
- Low density elastomeric insulation for the exit cone
- Aft-end toroidal igniter

GPS II PERIGEE INSERTION SOLID MOTOR STAGE

*STAR 48
TE-M-711-3
84-KS-15,000
UPPER STAGE MOTOR

PERFORMANCE CHARACTERISTICS @ 70°F AND VACUUM:
- AVERAGE THRUST 14,229 LB_F
- TOTAL IMPULSE 1,263,405 LB-SEC
- BURNING TIME 84.0 SECS.

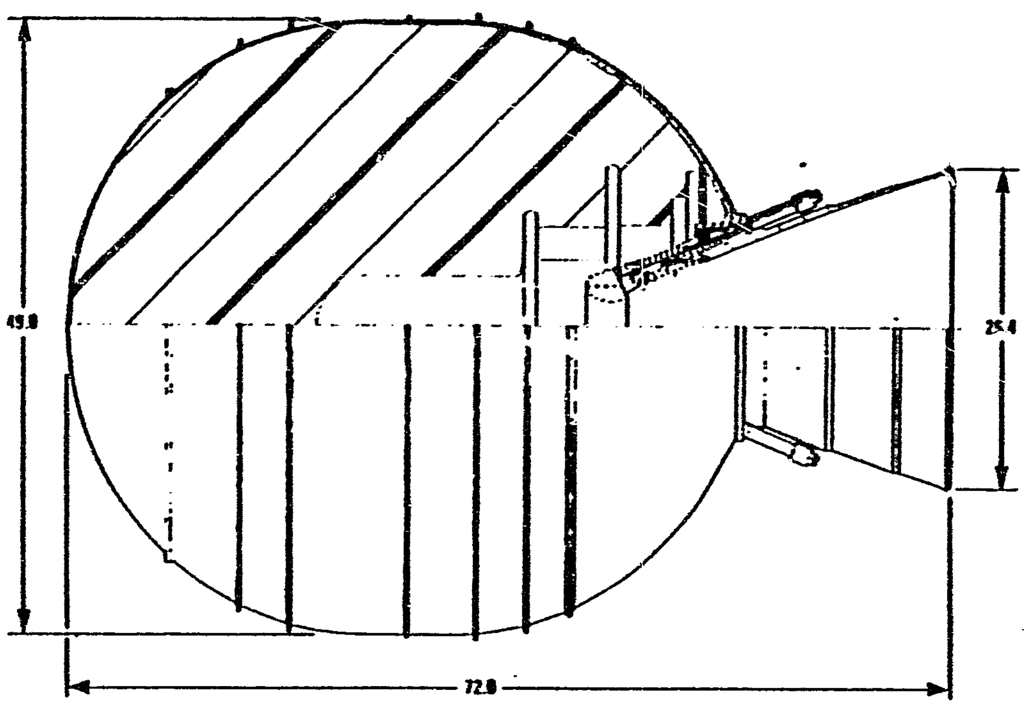
- SPECIFIC IMPULSE:
  - PROPELLANT 287 SECS.
  - EFFECTIVE 285.36 SECS.

PROP. WT.
4,401.5 LBM

LOADED MOTOR WT.
4,658.0 LBM

BURNOUT WT.
230.55 LBM

MASS FRACTION
0.945



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GPS II ORBIT INSERTION SUBSYSTEM DEFINITION

The Orbit Insertion Subsystem (OIS) motor planned for the GPS PHASE II design is the Thiokol STAR 37XF, shown in the facing pictorial. The motor was used for the first time in the recent launching of the Intelsat V.

The motor design incorporates many of the same features of the STAR 48, including 89% solid HTPB propellant in a high-strength, light weight 6AL-4V titanium case, a submerged nozzle with an integral toroidal igniter, and a carbon/carbon nozzle.

GPS II ORBIT INSERTION SUBSYSTEM

DESCRIPTION

EXISTING GPS BLOCK II MOTOR

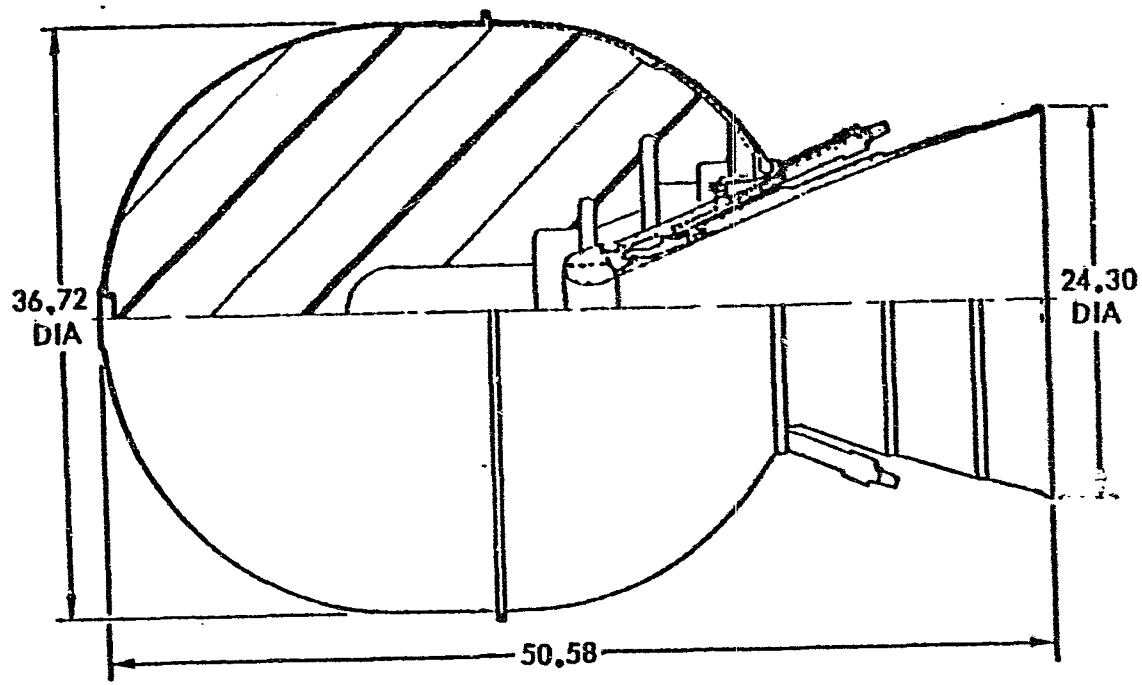
*STAR 37XF
TE-M-714-6
62-KS-9,000
APOGEE MOTOR

PERFORMANCE CHARACTERISTICS @ 70°F AND VACUUM:

- AVERAGE THRUST 10,000 LB_F
- TOTAL IMPULSE 556,000 LB-SEC
- MISSION OFF-LOAD 7%

LOADED MOTOR WEIGHT
2,077.5 LBM

EMPTY CASE WEIGHT
148 LBM



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P80-1 ATTITUDE CONTROL AND DETERMINATION SUBSYSTEM DEFINITION

The P80-1 Attitude Control and Determination Subsystem (ACDS) is illustrated in the facing pictorial. This is a 3-axis stabilized system from Shuttle Orbiter separation through all orbital operations. The primary attitude reference is the star sensor updated Inertial Measurement Unit (IMU). On orbit operations do not use the Reaction Control Subsystem, as attitude control is effected by the reaction wheels with electromagnetic dumping. The sun sensors are used for a sun-safe mode, and the Earth sensor for course acquisition. The normal on-orbit mode is zero yaw with 180° yaw flips when the sun line crosses the equator. The orbit altitude is 740 Km, which is in the TOPEX "ballpark."

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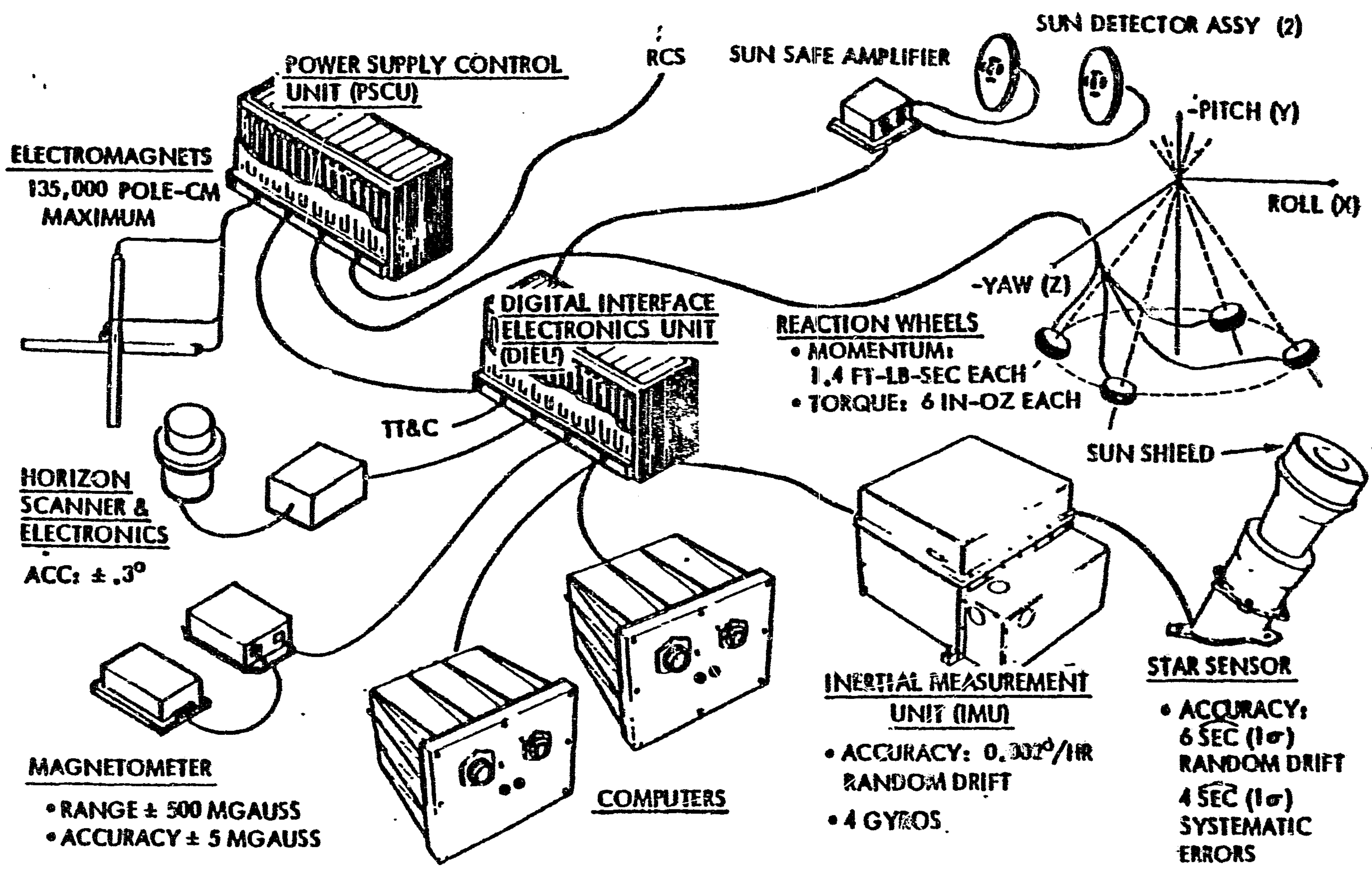
This system can be used for virtually any low altitude nadir pointing application with no significant changes. For inertially stabilized modes, intermittant maneuvers would be required to cause star transmits of the star tracker slits for IMU updates. In nadir pointing applications, the normal pitch rate would satisfy this requirement without additional maneuvering.

Total weight of the ACDS is 228.3 pounds (103.8 Kg) and it requires 134 watts of power.

The performance characteristics of the P80-1 ACDS is given in a table following on subsequent pages.

P80-1 ATTITUDE DETERMINATION & CONTROL SUBSYSTEM
DEFINITION



P80-1 ACDS ON-ORBIT PERFORMANCE DEFINITION

The P80-1 ACDS performance is summarized on the facing table. Only the nominal mode is of interest, in that the Teal Ruby telescope slewing introduces large disturbances that have to be allowed to damp out. Attitude determination is 0.0056° per axis (0.95P), much better than required. Attitude control of $\pm 0.1°$ in roll and yaw and $\pm .2°$ in pitch is relatively coarse, simply because there is no requirement to do any better. The reaction wheels are driven in a pulse width modulated mode with identical switching amplifier deadbands.

The performance could be improved to much less than 0.1° in all axes with relatively minor changes.

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P80-1 ACDS ON-ORBIT PERFORMANCE

FUNCTIONS	REQUIREMENT	PERFORMANCE
• ATTITUDE CONTROL		
NOMINAL	$\pm 0.7^\circ$ ALL AXES	$\leq 0.10^\circ$ ROLL, YAW $\leq 0.20^\circ$ PITCH
TRE OPS	$\pm 0.7^\circ$ ALL AXES DURING STARE MODE & STEPS OF 2.25° OR LESS	$\leq 0.6^\circ$ ALL AXES
	$\pm 0.7^\circ$ ALL AXES 30 SEC FOLLOWING SLEW $\leq 20^\circ$	ROLL, YAW 0.6° PITCH $\leq 0.7^\circ$ IN < 25 SEC
	$\pm 0.7^\circ$ ALL AXES 90 SEC FOLLOWING SLEW $> 20^\circ$	ROLL, YAW $\leq 0.6^\circ$ PITCH $\leq 0.7^\circ$ IN < 60 SEC
• RATE CONTROL		
NOMINAL	$\pm 0.01^\circ$/SEC ALL AXES	$\leq 0.005^\circ$/SEC
TRE OPS	PITCH $\leq 0.35^\circ$/SEC	$\leq 0.25^\circ$/SEC
	$\leq 0.01^\circ$/SEC ROLL, YAW 5 SEC AFTER STEP OF $\geq 2.25^\circ$	$\leq 0.01^\circ$/SEC IN ≤ 3.0 SEC
• ATTITUDE DETERMINATION	$\pm 0.0056^\circ$ (0.95P) EXCLUDES EPHEMERIS	$\leq 0.0056^\circ$
• RATE DETERMINATION	$\pm 0.75^\circ$/HR AT A DATA BANDWIDTH OF 5 Hz	$\leq 0.75^\circ$/HR DATA BW = 5 Hz
• DISTURBANCE TORQUES		
MAXIMUM (TRE)	1.25 FT-LBS	EXCEEDS SETTLING TIME REQUIREMENTS
	1.50 FT-LB-SEC	WHEELS SIZED FOR: PITCH 3.92 FT-LB-SEC ROLL, YAW 1.96 FT-LB-SEC

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GPS II ATTITUDE AND VELOCITY CONTROL SUBSYSTEM (AVCS) DEFINITION

The GPS Phase II attitude and Velocity Control Subsystem (AVCS) is illustrated in the facing pictorial. The subsystem was designed to operate in a 12-hour circular orbit pointing to nadir within $\pm 0.5^\circ$ and yawing about nadir continuously to maintain the solar arrays normal to the sun-line. There is no explicit yaw control during eclipse; yaw reacquisition is automatic at eclipse exit.

No Reaction Control Subsystem burns are used except for delta-V maneuvers. Momentum dumping is electro-magnetic, using stored Earth magnetic field data. On-orbit performance evaluation (of GPS Phase I spacecraft...from which this subsystem is derived) indicates that pitch and roll errors, as measured by the Earth sensor are less than $+0.1^\circ$. The specified accuracy of the unit is $+0.2^\circ$, (3ψ) although its performance has been determined to be $+0.1^\circ$ (3ψ).

The spacecraft is spin-stabilized in the transfer orbit with both active (prior to perigee burn) and passive nutation damping modes. The subsystem can be used with either a major or minor axis spinning vehicle. The rate gyro assembly is on board only for ground monitoring and is not used in any control mode (although it could be). The Control Electronics Assembly (CEA) for the GPS Phase II design is digital, in contrast to the GPS Phase I design, which is analog.

The weight of the AVCS is 87.9 pounds and it requires 25.1 watts of power.

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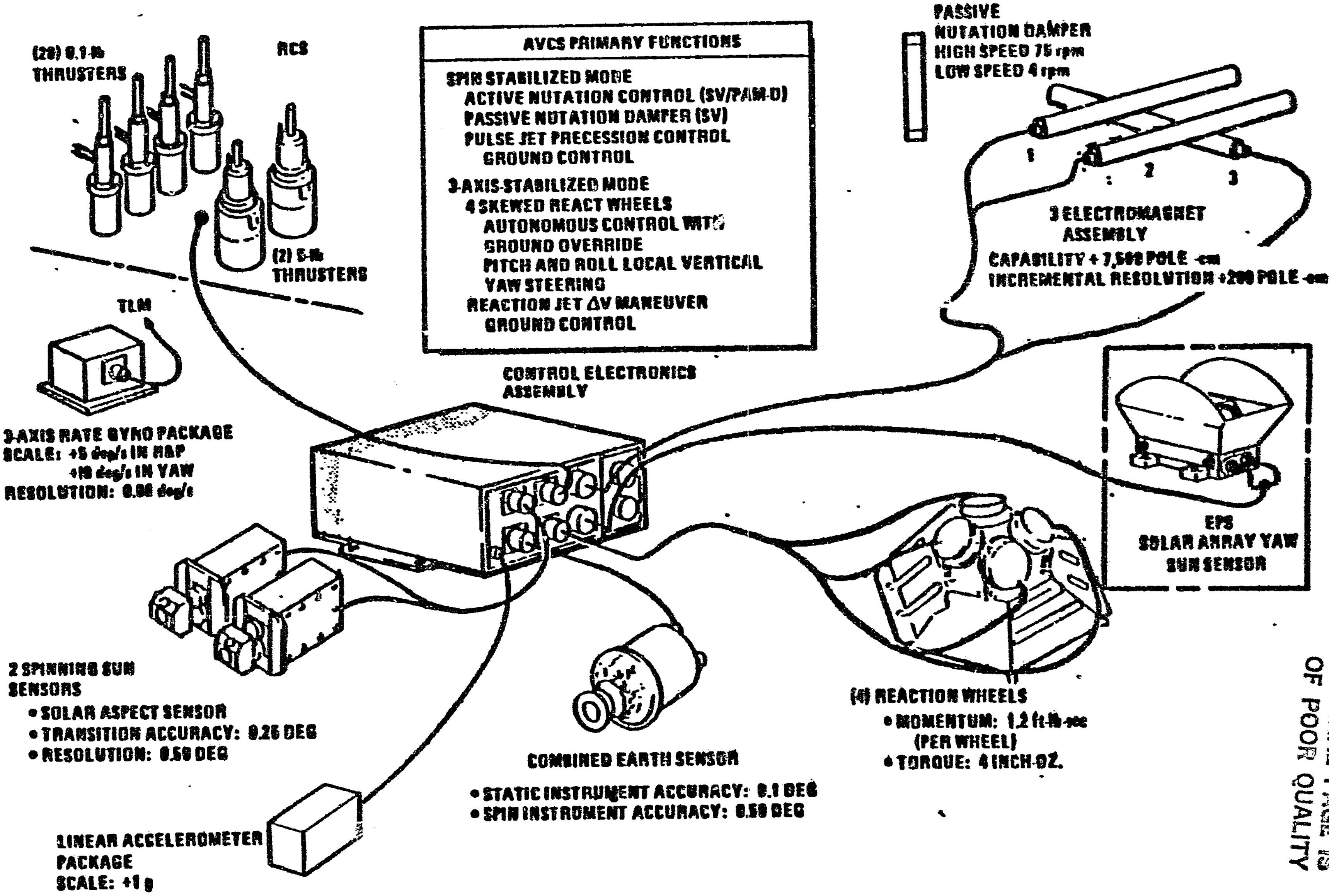
GPS II ATTITUDE AND VELOCITY CONTROL SUBSYSTEM DESCRIPTION



GPS II THERMAL CONTROL SUBSYSTEM (TCS) DEFINITION

In the on-orbit configuration, the GPS Phase II spacecraft is 3-axis stabilized so that the primary heat rejection surfaces are located on the shear panels and are always out of the sun. All of the external surfaces, except for radiators, are covered with multi-layer insulation (MLI). The shear panels are equipped with thermal louvers and a direct radiator for a high-power RF amplifier. Most of the equipment is mounted on an equipment shelf inside the spaceframe and operates continon an equipment shelf inside the spaceframe and operates continuously. Thermal doublers are installed on the equipment shelf to conduct heat to the shear panels. Thermostatically controlled heaters are installed on selected components to provide for mission cold phases.

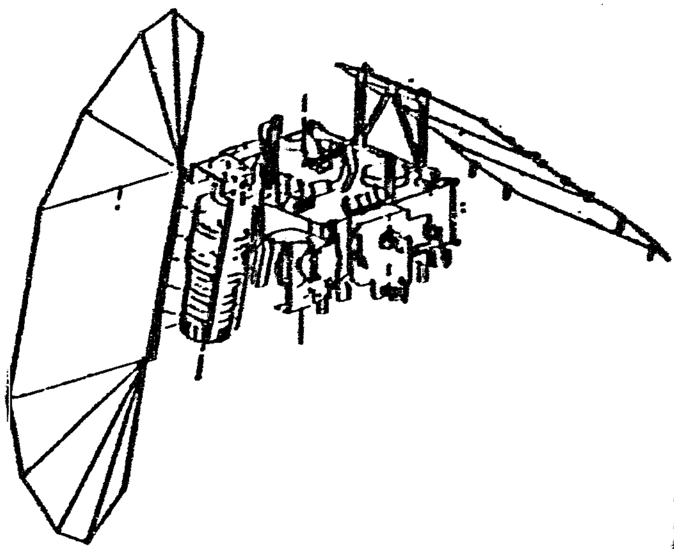
In the transfer orbit configuration, the spacecraft is in a spin mode with the solar arrays folded. This tends to balance solar insolation with the heat sink of deep space. The effects of apogee solid rocket motor burn plume heating are minimized by a heat shield, insulation blankets and a plume deflector. The effects of heat soak-back are minimized by thermally isolated motor mounts and insulation blankets.

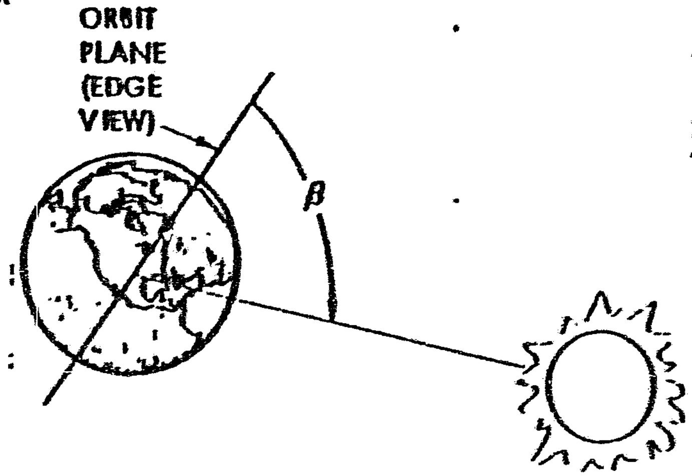
The design approach for the GPS II TCS is illustrated in the facing pictorial. The GPS II TCS weighs 142 pounds (64.55 Kg) and requires 27 watts of heater power intermittently.

P80-1 THERMAL CONTROL SUBSYSTEM MISSION DESCRIPTION

- ORBITAL PARAMETERS
  - PARKING: 160 NM CIRCULAR, 57° INCLINATION
  - TRANSFER: 160 X 400 NM, 63° INCLINATION
  - OPERATIONS: 400 + 50 NM CIRCULAR, 72.5° MIN INCL

- VEHICLE ORIENTATION
  - PARKING/TRANSFER (INERTIALLY STABILIZED)
    - IN ORBITER PAYLOAD BAY
      - DOORS OPEN 1 TO 3 HOURS AFTER LAUNCH
      - P80-1 EJECTED 5 HR 42 MIN AFTER LAUNCH
    - AFTER EJECTION FROM PAYLOAD BAY
      - THREE DAY STAY CAPABILITY WITH SOLAR VECTOR NORMAL TO SOLAR ARRAY
      - OTHER ORIENTATIONS BRIEFLY FOR BURNS
    - LAUNCH CONSTRAINT TO GIVE FULL SUNLIGHT ORBIT
      - BETA ANGLE BETWEEN 72.9° AND 80.5°
  - OPERATIONAL ORBIT (3-AXIS STABILIZED)
    - Z AXIS PARALLEL TO LOCAL VERTICAL
    - X AXIS PARALLEL TO VELOCITY VECTOR
    - BETA ANGLES FROM 0° TO 90°





P80-1 THERMAL CONTROL SUBSYSTEM DESIGN DEFINITION

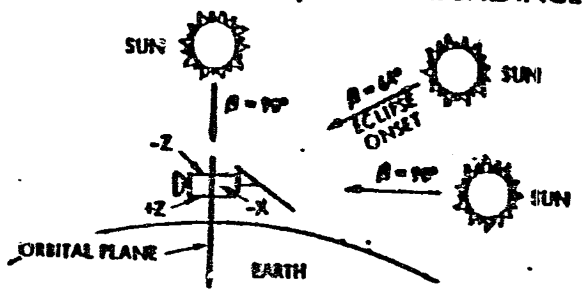
In the on-orbit configuration, the P80-1 spacecraft is in a 3-axis stabilized attitude such that the prime experiment (The Teal Ruby Steerable telescope) is located away from the sun. Since Teal Ruby uses the best location for heat rejection, the internal equipment radiators are placed on the Earth and anti-Earth facing panels. Low solar absorptive coatings are used to minimize the environmental heat load.

In the Shuttle Orbiter launch configuration, the P80-1 TCS is totally enabled as soon as the Mission Specialist enables the Electrical Power Subsystem. Analysis of 3 in-flight scenarios indicate that it takes approximately 3 hours before the on-board heaters are needed for the coldest environment. The Air Force has mandated that the P80-1 will be the first payload ejected from the manifest of that particular Shuttle flight. Therefore, no special Aerospace Support Equipment thermal shielding is required (as is the case for the GPS II). However, for other applications such thermal shielding could be provided.

The P80-1 TCS design rationale is presented on the facing page.

P80-1 THERMAL CONTROL SUBSYSTEM DESIGN APPROACH

- BASIC CONCEPT
  - COLD BIASED THERMAL CONTROL
  - MAKE UP HEAT
- CONCEPT RATIONALE
  - LOW COST AND HIGH RELIABILITY
- IMPLEMENTATION
  - RADIATORS INTEGRAL WITH STRUCTURE
  - MLI BLANKETS ON NON-RADIATOR SURFACES
  - THERMOSTATICALLY CONTROLLED HEATERS
- VERIFICATION
  - ANALYTICAL STUDIES
  - THERMAL VACUUM/THERMAL BALANCE TEST



P80-1 TCS WEIGHT: 55 LB.

HEATER POWER VARIES FROM 50 TO 100 WATTS, AND IS NOT CONTINUOUS

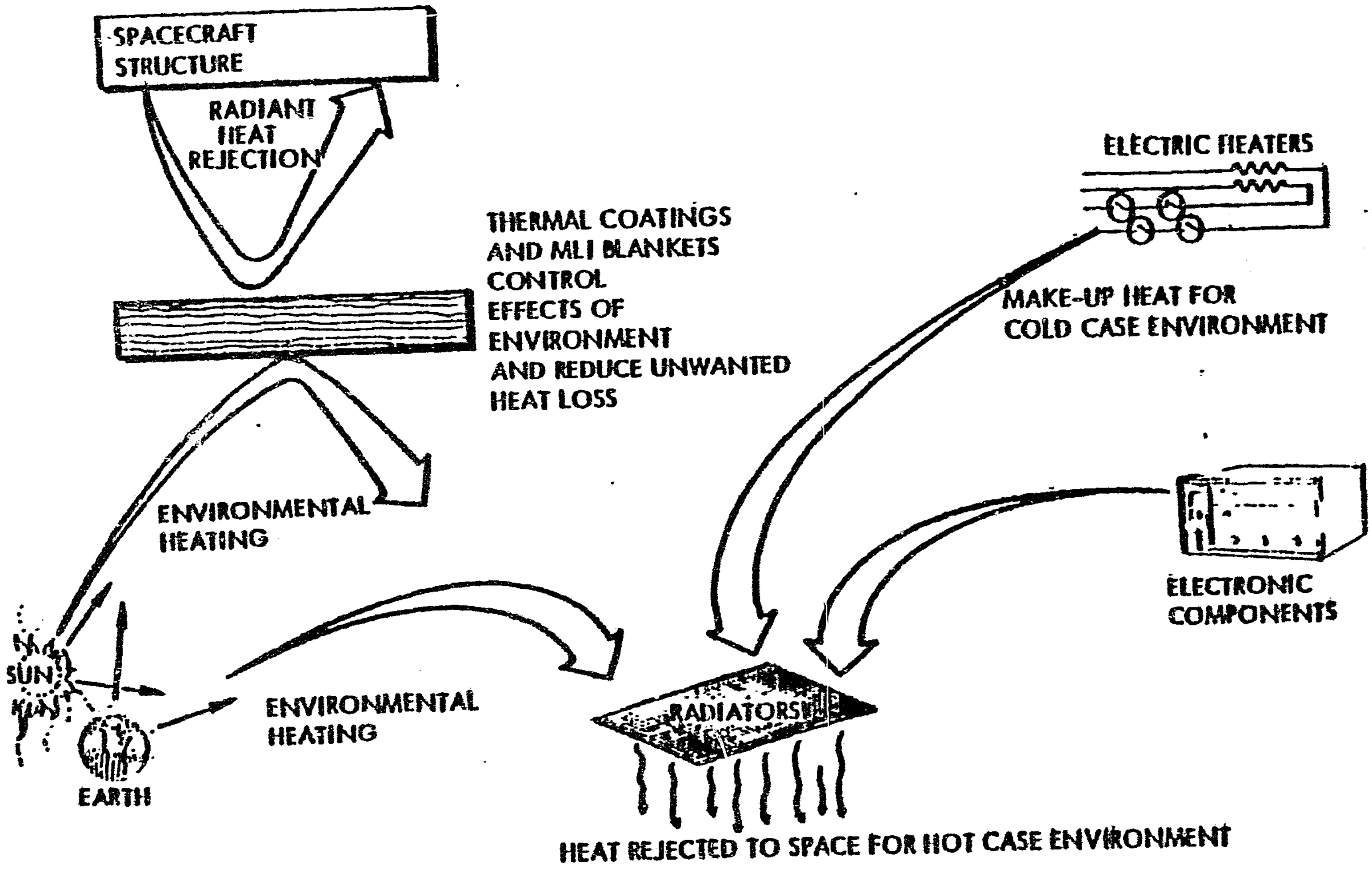
GPS II THERMAL CONTROL SUBSYSTEM (TCS) DEFINITION

In the on-orbit configuration, the GPS Phase II spacecraft is 3-axis stabilized so that the primary heat rejection surfaces are located on the shear panels and are always out of the sun. All of the external surfaces, except for radiators, are covered with multi-layer insulation (MLI). The shear panels are equipped with thermal louvers and a direct radiator for a high-power RF amplifier. Most of the equipment is mounted on an equipment shelf inside the spaceframe and operates continon an equipment shelf inside the spaceframe and operates continuously. Thermal doublers are installed on the equipment shelf to conduct heat to the shear panels. Thermostatically controlled heaters are installed on selected components to provide for mission cold phases.

In the transfer orbit configuration, the spacecraft is in a spin mode with the solar arrays folded. This tends to balance solar insolation with the heat sink of deep space. The effects of apogee solid rocket motor burn plume heating are minimized by a heat shield, insulation blankets and a plume deflector. The effects of heat soak-back are minimized by thermally isolated motor mounts and insulation blankets.

The design approach for the GPS II TCS is illustrated in the facing pictorial. The GPS II TCS weighs 142 pounds (64.55 Kg) and requires 27 watts of heater power intermittently.

GPS II THERMAL CONTROL SUBSYSTEM DESCRIPTION

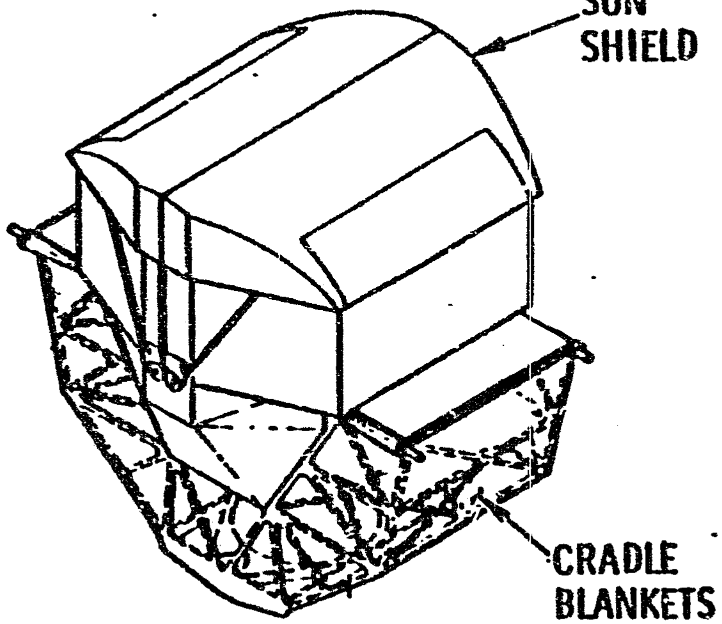


GPS II THERMAL CONSIDERATIONS IN THE LAUNCH CONFIGURATION

The GPS II differs from the P80-1 launch configuration in that as many as four (4) GPS II spacecraft may be carried to orbit in a single Shuttle flight. Therefore, some thermal protection is required after the cargo bay doors are opened and before all spacecraft can be separated from the Shuttle. Therefore, a sun-shade has bee provided which protects the spacecraft and the PAM-D perigee insertion system, as shown in the facing diagram. During the bay-door-closed phase (from launch to bay doors open)bay doors open), most components will remain within a few degrees of their launch temperatures. During the open-door phase, the thermal shade provides adequate thermal protection when the Shuttle cargo bay is Earth-facing. In a sun-facing attitude, some components will overheat within about 30 minutes. In a space-facing attitude, heating of some selected components will be required in about 90 minutes.

As illustrated, the sun-shade (furnished as part of the PAM-D/ASE system) is of a clam-shel design and has its own self-contained electric motor for retraction. MLI thermal blankets are provided to protect the heat from leaking out the bottom of the launch cradle.

GPS II THERMAL CONSIDERATIONS IN THE SHUTTLE CARGO BAY



SUNSHIELD CLOSED

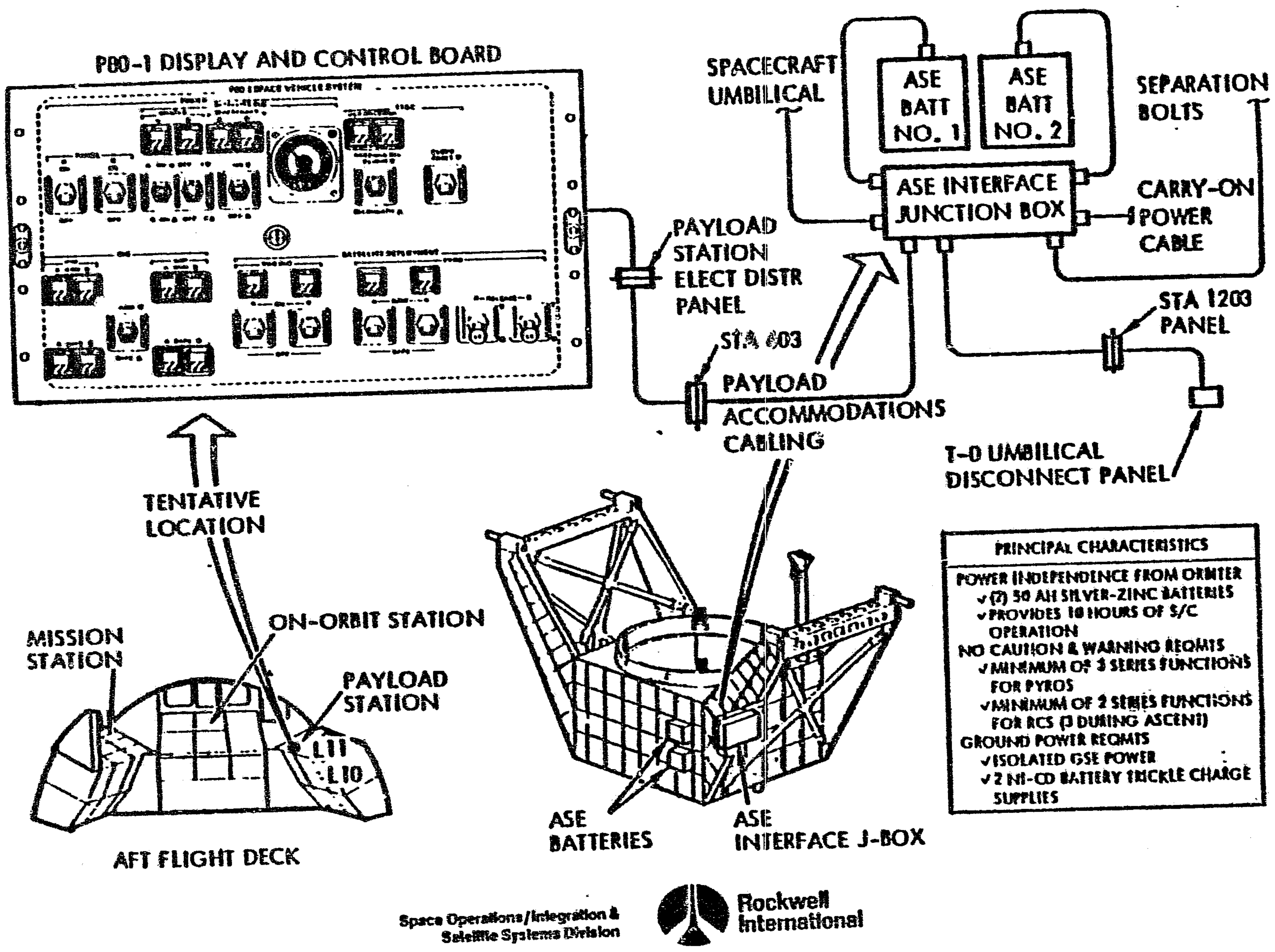
P80-1 AEROSPACE SUPPORT EQUIPMENT (ASE)

Descriptions and performance characteristics of the P80-1 and GPS Phase II Shuttle Orbiter cargo bay and aft flight deck Aerospace Support Equipments (ASE) has been included in this task.

The P80-1 ASE is composed of the Launch Cradle and the Display and Control Panel, as shown in the facing chart. These items were designed and fabricated by Rockwell specifically for the Shuttle Launch of P80-1, but are adaptable for the launch and Shuttle cargo bay separation of similar spacecraft.

The launch cradle is designed to protect the payload (spacecraft) from Shuttle Orbiter launch environments. A Pyrotechnic/separation spring arrangement is used to separate the spacecraft, but does not provide a spin table for spin-stabilized spacecraft; (P80-1 is a three axis control system). The launch cradle weighs 1,949 lbs., and interfaces with the perigee kick stage (Thiokol TE-M-364-4) for retention of the payload from launch to separation. Two (2) 50 A-H silver-zinc batteries are mounted to the cradle structure and used to supply spacecraft power for pre-launch and in-bay checkout prior to separation. An ASE junction box is also attached to the cradle (see diagram). Power from the batteries is used to initiate the pyrotechnically operated separation bolts at the cradle side of the interface.

Cabling from the ASE junction box is routed to the shirt-sleeve environment aft flight deck of the Shuttle Orbiter, where a mission/payload specialist can control the spacecraft separation after voice communication coordination is received from the ground. Isolated ground power provisions are made for pad operation prior to lift-off, which is also used to trickle charge the spacecraft batteries. No caution or warning provisions are supplied because each pyro function, (and enablement of the RCS engines during ascent) requires three (3) series operations. The cabling terminates at Station 603 bulkhead connector, and also has a Payload Station disconnect inside the aft flight deck, All electrical (and mechanical items other than the cradle itself) weigh an additional 168 lbs. for a total ASE weight of 2,117 lbs. The ASE is designed to meet all of the safety requirements of NASA Handbook 1700.7A and includes the capability to safe all systems for return from orbit in the event separation could not take place.



P80-1 SHUTTLE LAUNCH AEROSPACE SUPPORT EQUIPMENT

GPS PHASE II AEROSPACE SUPPORT EQUIPMENT (ASE)

The Shuttle cargo bay GPS Phase II ASE is supplied as a package by the MacDonald-Douglas Corp., and includes the basic launch cradle, a payload spin-up table (the GPS II is spin-stabilized during orbit insertion) which is capable of de-spinning the payload in the event separation cannot be accomplished, and a sun shield/cradle thermal blanket for solar protection in the cargo bay prior to payload separation. This was deemed necessary in that as many as four (4) spacecraft may be launched on the same mission, and the time delay in the sun, from bay doors open to separation of the last payload, may cause overheating of the spacecraft. Use of the sunshield for TOPEX would require further study and some knowledge as to the time when the TOPEX spacecraft would be separated after the bay doors were opened. Total weight of all in-bay ASE items is 2,515 lbs., and includes the Payload Attach Fitting (PAF), the Launch Cradle, Spin-Table, Electronics (attached to the cradle), and the thermal items (sunshield and blankets). The thermal items are designed to protect the dormant payload in the Shuttle cargo bay from the following environments:

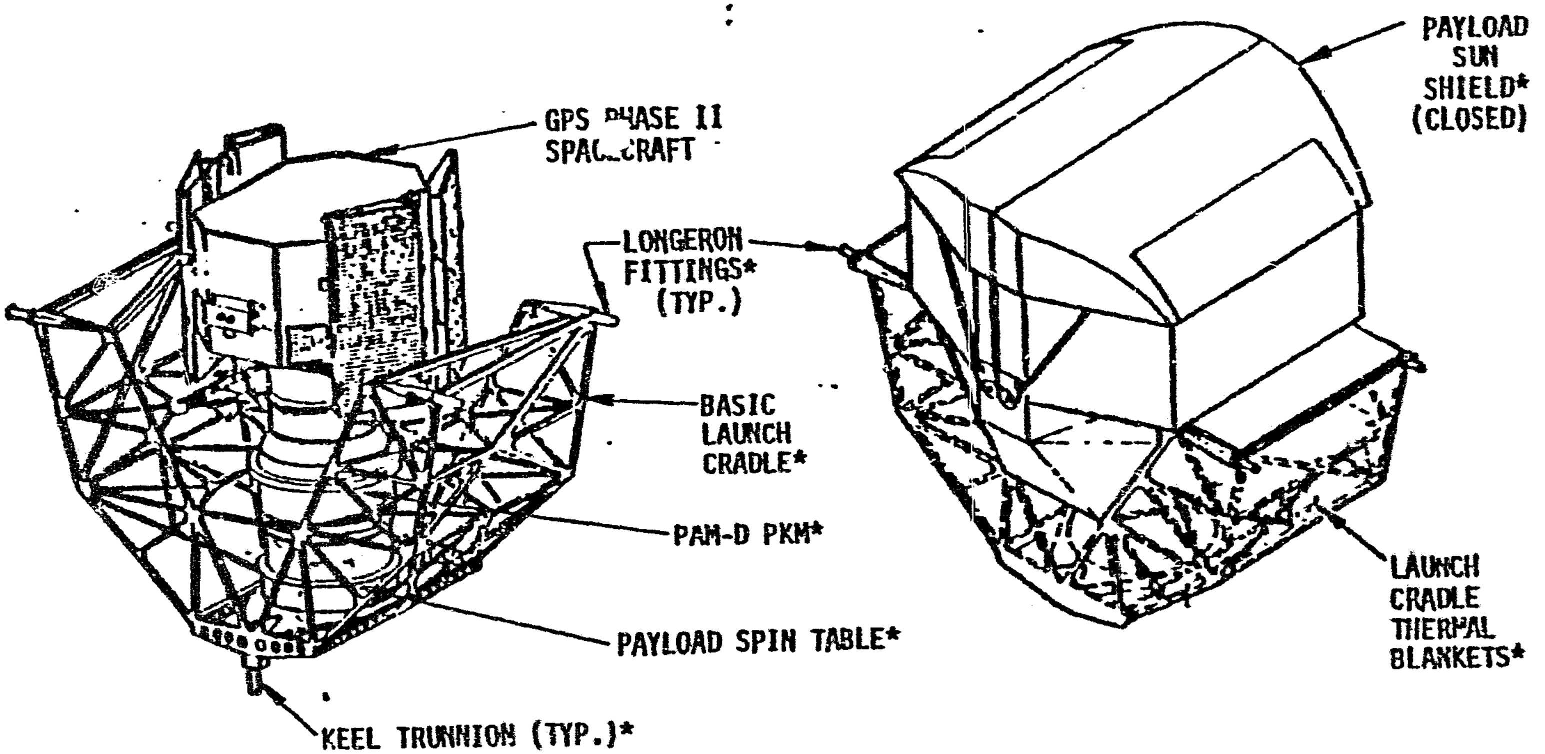
Condition	Minimum	Maximum
1. Prelaunch	+40°F (4.4°C)	+120°F (48.9°C)
2. Launch	+40°F (4.4°C)	+150°F (65.6°C)
3. On-orbit (doors open)	-250°F (-156.7°C)	+200°F (93.3°C)
4. Entry and postlanding	-50°F (-45.6°C)	+200°F (93.3°C)

Conditions 1 and 2 for assumed adiabatic payload

Condition 3 for assumed empty cargo bay

Condition 4 for assumed adiabatic payload. Maximum temperature is for assumed initial 70°F cargo bay wall temperature. Minimum temperature is for assumed initial -250°F cargo bay wall temperature. Local areas around vents may reach 225°F for less than 2 minutes after vents are opened.

The controls required in the Aft Flight Deck for manual initition of spin-up and separation of the GPS II is described on the following charts.



*SUPPLIED AS A PACKAGE BY THE MACDONALD-DOUGLAS CORPORATION.

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Figure GPS PHASE II SHUTTLE LAUNCH AEROSPACE SUPPORT EQUIPMENT

Space Operations/Integration & Satellite Systems Division

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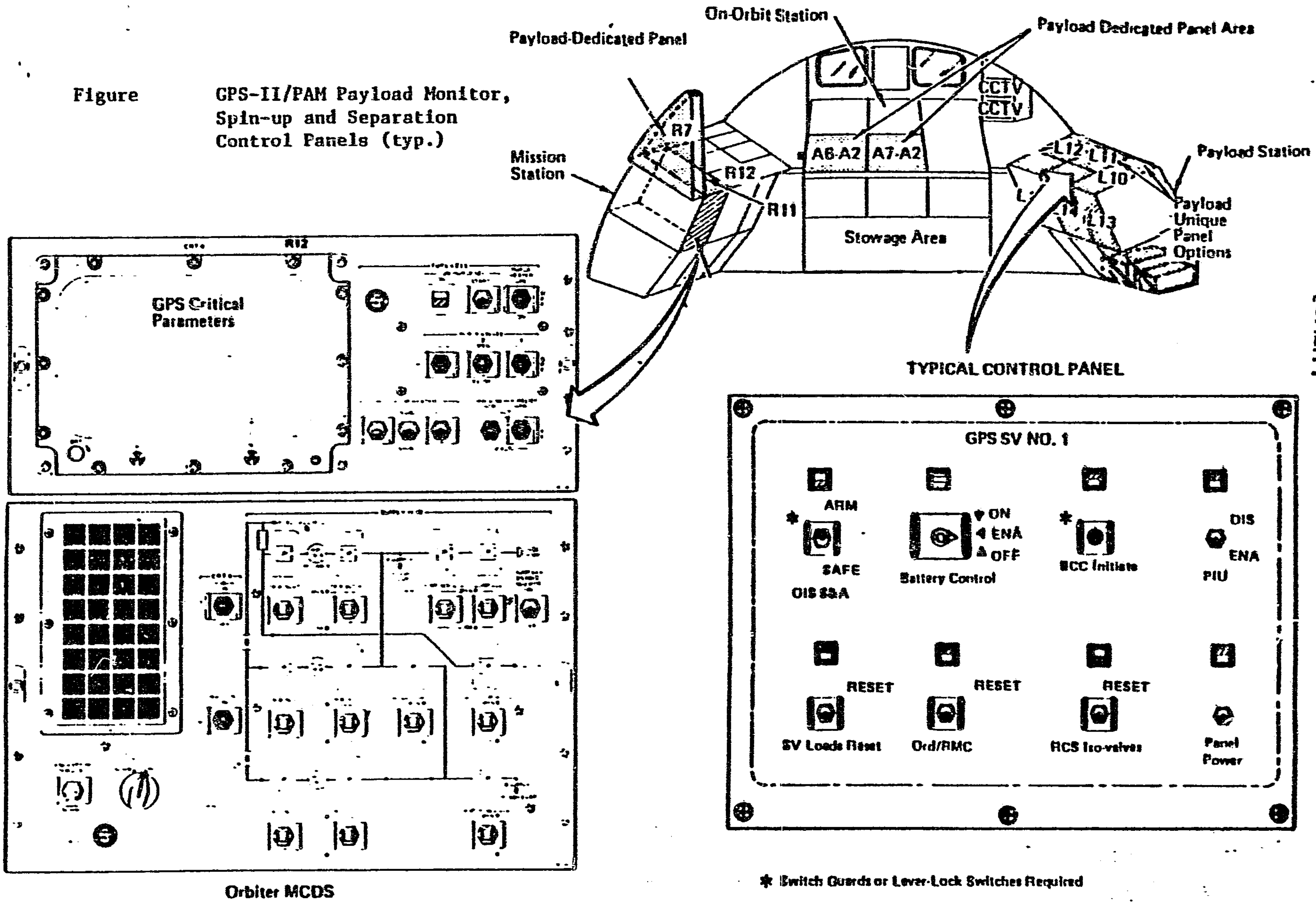
GPS PHASE II AFT FLIGHT DECK AEROSPACE SUPPORT EQUIPMENT

The checkout and separation of the GPS Phase II spacecraft and the upper-stage Payload Assist Module-D (PAM-D) is a little more complex than the equivalent for P80-1 because the GPS/PAM stack requires spin-up prior to seperation (GPS Phase II is spin stabilized through orbit insertion and circularization). The exact interface design has not, as yet been firmed but will look very much like that shown in the facing diagram. The integrated aft flight deck crew station is arranged for checkout, monitor, command and control of the cargo operations with the Multi-function computer Display System (MCDS) used in conjunction with the GPS and the PAM to monitor and limit-check critical parameters. An aft flight deck switch panel design concept is shown on the facing diagram which will accommodate from one to four GPS/PAM configurations and allows the flexibility to launch with any shared payloads without an electronics interface constraint which would limit the GPS availability.

The aft flight deck Aerospace Support Equipment (payload chargeable) would weigh approximately 40 lbs., and contain all the safety features and constraints required by the NASA Handbook 1700.7A, including the safing requirements for return to Earth from orbit in the event the payload could not be separated.

Figure GPS-II/PAM Payload Monitor, Spin-up and Separation Control Panels (typ.)



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TEST, GROUND SUPPORT EQUIPMENT AND SOFTWARE

This section explores the ramifications of qualification/acceptance testing of new and modified hardware for both P80-1 and GPS II subsystems, and the system qualification/acceptance for the built-up spacecraft.

A discussion of the Ground Support Equipment (GSE) availability for each spacecraft is included, as well as the required software changes and availability.

P80-1/GPS TOPEX Test Program

TOPEX Payload Components - It is assumed that all of the GFE payload components will have been qualified for flight on the Shuttle Orbiter and that the flight units will have been acceptance tested prior to delivery to the contractor for installation on the spacecraft.

"Reused" Subsystem Components - All spacecraft subsystem components which are "carried over" from either the P80-1 or GPS II without modification will have been qualified by the requirements of the very stringent MIL-STD-1540A to the environmental levels of the Shuttle Orbiter. Flight units will be acceptance tested by the supplier prior to delivery. Any component which must be modified will be re-qualified to a level depending upon the degree of modification required. Slight modification may result in only a "partial" qual, whereas a complete rework may require a requalification as though it were a new item.

"New" Subsystem Components - Any new component required to meet TOPEX mission requirements can be qualified in a number of different ways, (1) full environmental qualification (which requires either a dedicated qualification unit, or that it be completely refurnibhsed and acceptance tested prior to being considered flight-worthy); (2) "protoflight" qualification, which is subjecting the unit to an environmental qualifi-cation level that is somewhat less than "full" qualification, but sufficiently high to assure spaceflight integrity; such a qualification test does not require refurbishment prior to flight; (3) qualification by "similarity" which can be used for components that are new to P80-1 or GPS II, but which have been previously qualified for other programs at levels that are as high, or higher, than that which they will see in this application. This requires certification by the supplier and agreement by the customer, but does not require any actual testing. Note that all units to be qualified must be first acceptance tested. Re-acceptance testing is not required after protoflight testing. All units scheduled for flight but not requiring qualification must be acceptance tested prior to flight.

Spacecraft System Testing - Although both P80-1 and GPS II will have been qualified at the system level, their TOPEX configuration will involve some different distributed weights, different payload, etc., and will require some qualification exposure at the system level. "Protoflight" testing can also be used in this instance as a cost effective measure. As an example: Full-qualification in acoustics (per MIL-STD-1540A) requires +6 dB above expected OASPL for a period of three minutes. A protoflight qualification program may require only +3 dB for two minutes (acceptance testing would require 0 dB for one minute).

As a minimum, the following system-level tests will be recommended:

- Acoustic vibration
- Thermal balance/thermal vacuum
- EMC
- Live-pyro shock, including cradle separation tests
- Functional (foreshortened) prior to and after each environmental exposure. Proof pressure and leak tests are conducted after RCS installation in the spaceframe, but before installation of any electronics. Leak checks are repeated after all dynamic tests and before shipment for launch. Spin-balancing will be required of a GPS II TOPEX satellite spacecraft.

P80-1 Ground Support Equipment - As P80-1 is a "one-shot" spacecraft program, it is assumed that its GSE will be stored after the space craft is launched and will therefore be available for TOPEX when required (i.e., "housekeeping subsystem GSE"). It is further assumed that all ground support equipment for checkout of the TOPEX payload will be delivered as government-furnished equipment.

<u>P80-1 Software</u> - Same as GFE. The telemetry and command software from P80-1 will probably be useless, because of the change from Air Force TT&C to the NASA C&DH/TDRS telecommunications subsystems; and new software will have to be procured. All other software required will be available, although it might require modification.*

<u>GPS II Ground Support Equipment</u> - The GPS II production line will be active at the time of the TOPEX go-ahead date. All existing GSE for production testing will probably be in full use. Thus, a complete set of GPS II GSE will have to be procured for TOPEX use (except for GPS II payload GSE).

<u>GPS II Software</u> - Same as P80-1 (with respect to telemetry and command software). However, housekeeping subsystem checkout may require procuring an additional set which is not a large cost item. The Attitude and Velocity Control subsystem's Control Electronics Assembly software will have to be procured new because of yaw determination requirements and low Earth orbit application.

*It is assumed that for both P80-1 and GPS II TOPEX versions, the software required to checkout and reduce the data from the TOPEX payload will be government furnished.

Additional Information

Included in this section is a brief preliminary description of the TOPEX mission sequence (applicable to both P80-1 and GPS II) and a preliminary schedule.

PRELIMINARY NOMINAL TOPEX MISSION SEQUENCE

The nominal Topex mission sequence is shown on the facing diagram. It is a bit arbitrary, but does give a feel for the mission. Although the mission sequence shown was derived from P80-1 data, it shows a Topex mission which would also apply to GPS II.

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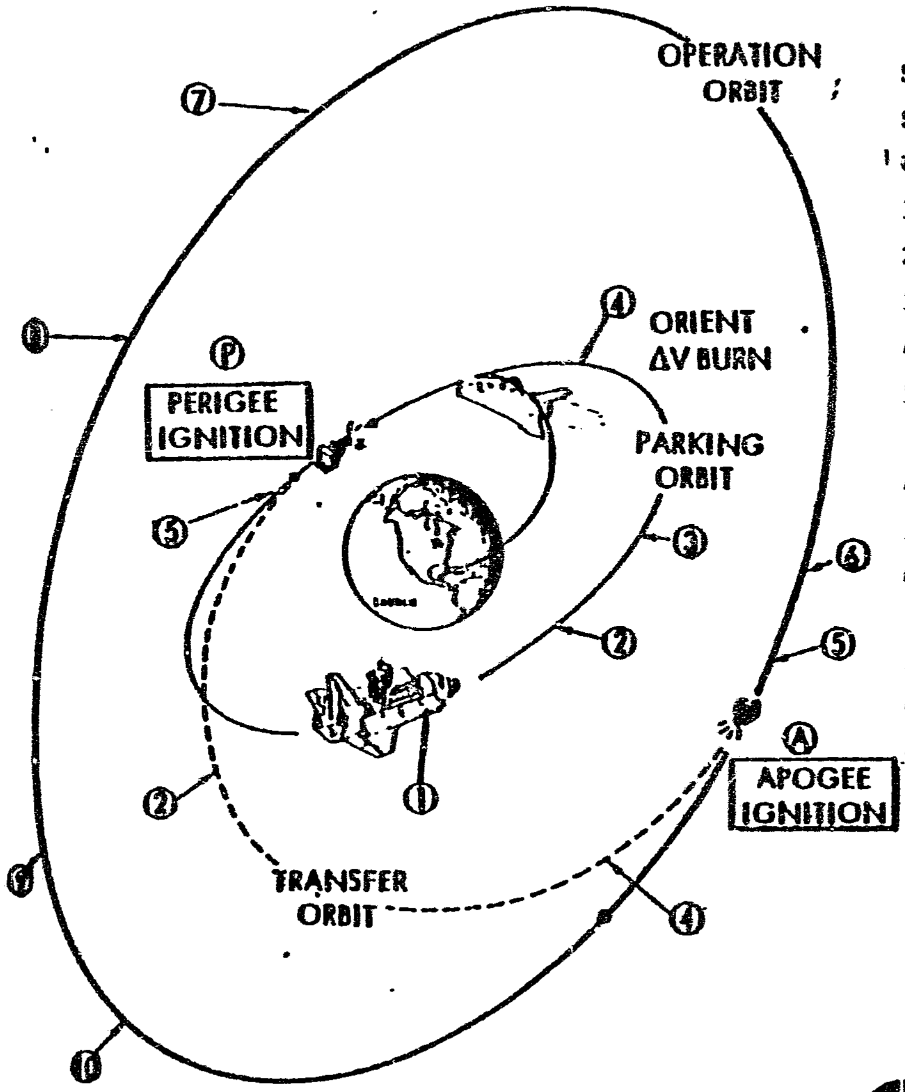
PRELIMINARY NOMINAL MISSION SEQUENCE



EVENT	TIME FROM LAUNCH HR MINS
SHUTTLE ORBITER LAUNCH	0 0
SHUTTLE ORBIT INSERTION	0 46
CONFIGURE & READINESS TEST SPACECRAFT	4 0
1. SEPARATE SPACECRAFT	5 42
2. SOLAR ARRAY/SUN ORIENTATION (IF REQ'D)	6 0
3. ORIENTATION COMPLETE	6 5
4. RE-ORIENT FOR PERIGEE INSERTION	6 25
5. INITIATE PERIGEE DELTA-V BURN (P)	6 42
-RE-ORIENT FOR SOLAR ARRAY/SUN (IF REQ'D)	6 50
4. RE-ORIENT FOR APOGEE INSERTION	7 10
5. INITIATE APOGEE DELTA-V BURN (A)	7 30
6. DE-SPIN, DEPLOY, ON-ORBIT CONFIGURATION	7 45
7. PERFORM ORBIT TRIM	14 55
8. ORIENT EXPERIMENT POINTING TO NADIR	17 56
9. PAYLOAD MISSION INITIATED	18 24
0. PERFORM SMALL DELTA-V MANEUVERS	AS REQ'D.

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CONCLUSIONS AND RECOMMENDATIONS

This section summarizes the conclusions that can be derived from the study and makes some recommendations based on the results of the study.

CONCLUSIONS:

1. Either the P80-1 or the GPS Phase II spacecraft buses can be used to satisfy the TOPEX mission requirements with the given TOPEX payload.
2. Off-the-shelf propulsion hardware exists for perigee and apogee insertion uses (i.e., solid rocket motors).
3. Neither the P80-1 nor the GPS II Telemetry, Tracking and Command subsystems lend themselves, even with modification, to the TOPEX mission and must be replaced with a NASA compatible C&DH/TDRS subsystem. Such hardware exists and could meet TOPEX requirements.
4. The Electrical Power subsystems of both spacecraft will meet the payload demand of the TOPEX payload, provided the empty solar array areas of GPS II are filled with solar cells.
5. No major modifications of either spacecraft is required to physically accommodate the TOPEX payload with the exception of the Option 1 two-meter radio altimeter antenna on the GPS II. Alternate antenna designs could be used.
6. The P80-1 Attitude Control and Determination subsystem could be used for the TOPEX mission without modification; however, the subsystem is over-designed for TOPEX use and deletion of unnecessary components and other simplifications could be cost effective and more power efficient.

CONCLUSIONS (Cont.):

7. The GPS II Attitude & Velocity Control subsystem would require modifications for low Earth orbit use and TDRS antenna pointing integration.

8. The Reaction Control subsystems of both spacecraft can be used without modification (including use of existing expendable amounts) to meet TOPEX three-year mission requirements. P80-1 would require some modification to meet the two-year extended mission (five-year mission) option.

9. The Thermal Control subsystems of both spacecraft can accommodate the TOPEX payload and mission with passive techniques and would not require major modification to meet TOPEX requirements.

10. The P80-1 configuration in its launch cradle provides flexibility with little modification to adapt to various perigee and apogee insertion stages (solid rocket motors). Variation of such stages for GPS II would be a little more complex to adapt to an in-bay spin table.

RECOMMENDATIONS:

Select either the P80-1 or GPS II design and authorize an in-depth study, with sufficient resources to allow a point-design concept to be developed for the TOPEX mission.